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Helicopter Aeromechanics

NORTH ATLANTIC TREATY ORGANIZATION

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ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT

(ORGANISATION DU TRAITE DE L'ATLANTIQUE NORD)

AGARD Lecture Series No.139

HELICOPTER AEROMECHANICS

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- Improving the co-operation among member nations in aerospace research and development;
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HELICOPTER AEROMECHANICS
INTRODUCTION AND HISTORICAL REVIEW

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SUMMARY

Although the current generation of civil and military helicopters have greatly improved upon the previous generations, rotorcraft technology still has the potential for decided improvements. As high-payoff technologies aeromechanics (aerodynamics/dynamics), structures/materials, avioncis/flight controls, and engine drive system technology can be identified, of which aeromechanics will be discussed in more detail. The general role of aeromechanics in military and civilian applications to rotary wing aircraft will be summarized and an overview of the state of the art will be given.

In addition to direct performance improvements, there is great opportunity to improve the operating characteristics to a degree that the full performance characteristics inherent in the designs may be realized. Some current systems fail in the utilization of their full performance as a result of dynamical limitations or high vibrational levels.

Integrating the research and technology opportunities with advanced rotorcraft configurations will yield promising new vehicles for civilian and military applications.

PREFACE

In 1973, AGARD Lecture Series "Helicopter Aerodynamics and Dynamics"[1] co-sponsored by the Fluid Dynamics Panel and the Consultant and Exchange Program of AGARD was presented. This lecture series had been conceived to provide an extensive overview of the role of aerodynamics and dynamics in the helicopter development from the fundamental methods and principles, through conceptual design to flight test and proof-of-concept. At that time only relatively old textbooks existed in this field, and the series had been structured to meet what was considered to be a deficiency in the literature. The compilation of the lectures provided a worthwile publication to accomplish this objective.

Now, more than ten years later, the overall situation concerning helicopter literature has substantially improved. The excellent textbooks by Bramwell[2], Johnson[3], and Stepniewski,Keys[4] satisfy a long-felt need. They cover the fundamentals of helicopter theory as well as the methods of engineering analysis required to design, develop, and evaluate helicopters and give a broad review on advanced subjects, and recent accomplishments. Therefore, a new lecture series on helicopter aeromechanics can be focussed on the latest technological developments, and achievements in helicopter aeromechanics and the modern trends covering the influence on performance and flight characteristics.

It is considered more important to provide a general appreciation of the phenomena and to discuss the state of the art of knowledge and methods as well as new contributions to modern technology, than to report on the details of the latest achievements in various research areas. For these details reference has to be given to the numerous special publications, mainly in the proceedings of the American Helicopter Society Annual Forum and the European Rotorcraft Forum, the helicopter journals, and the AGARD reports on some specialists' meetings covering helicopter related subjects[5-10]. For additional information a bibliography prepared specially for the purpose of this lecture series and restricted to material published within the last five years has been included as supplement in the publication of the lecture series.

1. THE ROLE OF AEROMECHANICS IN ROTARY WING AIRCRAFT FOR MILITARY APPLICATIONS

1.1 General Aspects and Historical Review

Aerodynamics, flight mechanics, dynamics, aeroelasticity, and acoustics together form what is designated as aeromechanics. There is no doubt that aeromechanics plays a key role in the meeting of requirements for application to military and civil use of rotary wing aircraft. Aeromechanics is central in its importance to future progress in rotorcraft.

This introduction will provide insights into the complex situation of helicopter design and operation and the technologies involved, starting from the first helicopters up to modern and future helicopters. Of course, there are several excellent review papers in this area covering the role of aeromechanics in military and civilian applications of rotary wing aircraft, for instance Yaggy[11] or Augustine and Morrison[12], and special papers deal with the needs, opportunities, and benefits of future rotorcraft, for instance[13]. In parts this paper will follow these other contributions, but by plagerizing freely a somewhat different perspective is intented.

The past is reviewed, only to such an extent necessary to understand the special problems in the development of the helicopter and its slow progress compared to the fixed-wing aircraft, roughly following an older historical review[14]. The concept of using air screws or rotors for vertical lift and vertical flight is very old, and the first flight trials of helicopters were about the same time as the first flights of fixed wing aircrafts. But these early attempts with rotary wing aircraft were without success.

The aerodynamic flow situation of a helicopter rotor is determined by the rotation of the rotor. In forward flight the speed of flight will be superimposed on the rotational velocity, resulting in periodically changing velocities for the blades with opposite characteristics at the advancing and at the retreating side. That means, that a helicopter, even in steady flight condition, is submitted to unsteady rotor conditions. The periodically changing velocities cause alternating forces and moments at the blades and at the rotor hub resulting in many dynamic effects which determine the special characteristics of a helicopter. The historical story of the helicopter is marked by the struggle against the dynamic forces and moments and the dynamic characteristics of the whole helicopter. This struggle was for a long time unsuccessful for several reasons. But mainly, there was not sufficient understanding of the aeromechanical and structural problems and no solutions to overcome these problems.

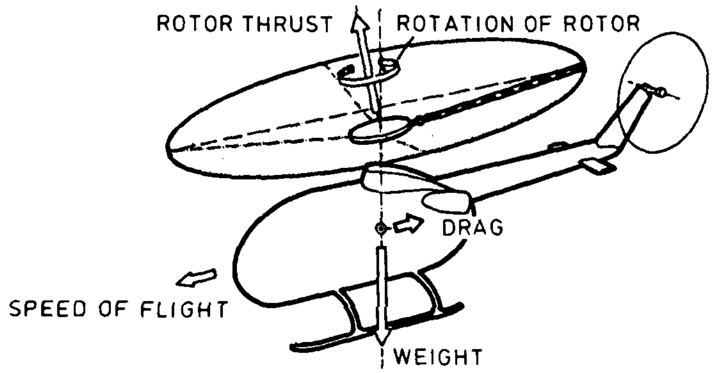

Figure 1 Helicopter in forward flight

The predominating opinion in this very early era was that there would be no future for rotary wing aircraft for technical reasons, and that it would make no sense to work in this field. In the following years engineering effort concentrated on developing very successfully the fixed-wing aircrafts, but men were aware that they still had to achieve complete mastery of the air; namely, the ability to stay aloft without maintaining forward speed and to ascend and land vertically in restricted areas. The development of the helicopter continued to this end.

The most pivotal effort in the attempt to achieve successful flying of rotary wing aircraft was the work of Juan de la Cierva in the 1920's in developing the first truly successful rotary wing aircraft which he called the "Autogiro" (Figure 2). This aircraft used a propeller for forward motion, and a freely rotating rotor for lift. The Autogiro did not actually achieve truly vertical flight; it required small forward speeds to maintain its lift. Autogyro development continued in Europe and in America for more than a decade and reached a state of considerable advancement, but there was no commercial break through and the progress came almost to a stand still. There was not enough potential for military application because of its imperfection in the vertical lift capability.

Figure 2 Cierva - Autogiro C19 Mk IV

However, the autogyro development laid the groundwork necessary for practical helicopter flight by providing knowledge and technology. It was only with the introduction of the articulated blade attachment done by de la Cierva that the prerequisites for realization of technically satisfactory helicopter projects were created. At that time, with the insufficient knowledge of the physical-technical correlations the blade attachment hinges were the only way to overcome the structural problems caused by high alternating loads at the blade root and the control problems of the whole helicopter caused by the dissymmetry of the flow field on the two sides of the rotor disc. All helicopters of the following period and still most of today's helicopters have articulated blade attachment in one of the many possible variants.

Following autogyro development, work advanced toward successful helicopters. In 1936, Focke in Germany demonstrated a side-by-side, two-rotor machine (Figure 3), and in 1939-1940 Sikorsky in U.S.A. introduced the VS-300 (Figure 4), a single lifting rotor machine with a vertical tailrotor for torque counteraction. Both of these designs are the real beginning of the practical helicopter era. They gave tremendous impetus to further development. In the late 1940's, the general pattern of helicopter-type aircraft had been formulated to a rather complete degree and nearly all of the current configurations had been given serious consideration by the 1950's, including single rotor, tandem rotors, coaxial rotors, side-by-side rotors, shaft driven and tip driven rotors and compound rotor concepts.

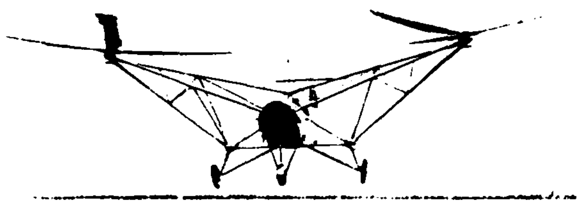

Figure 3 Focke Helicopter FW 61

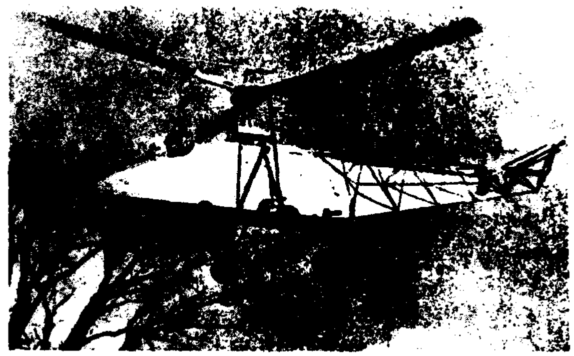

Figure 4 Sikorsky Helicopter VS-300

In the more recent era, general advance in helicopter knowledge - in its flight mechanics, its dynamics and structural properties - has progressed and designers returned to concepts with rigid blade attachment as in the beginning of the helicopter history. With the so-called hingeless rotor systems the flapping and lagging motions of the rotor blades, prevailing in articulated rotors, are replaced by elastic blade deformations. Such rotor systems have been made possible by the introduction of fiberglass composite rotorblades, for which MBB did the pioneering work in the late 1950's and early 1960's. Only the blade torsional or feathering hinges for blade control are provided with the hingeless rotors. In the more advanced so-called bearingless rotor systems these hinges are replaced as well by elastic motions. Figure 5 illustrates the mechanical simplification achieved by elimination of blade attachment hinges which is possible with different rotorsystems. Today, there is a broad variety of good rotor designs, also including composite hubs and so-called elastomeric bearings.

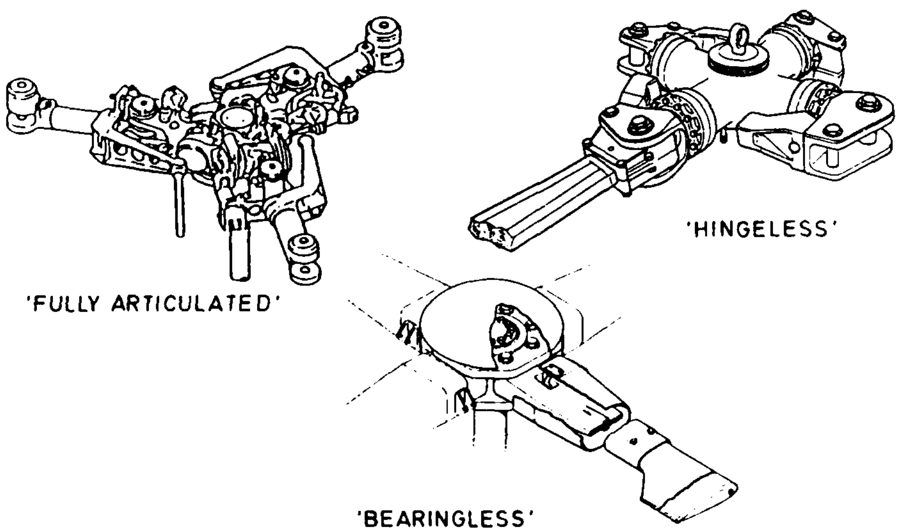

Figure 5 Different Rotorsystems

There is often discussion about the maturity of the rotary wing aircraft. Some people say that helicopters " are some forty years behind fixed wing aircraft in terms of total flight experience,... that rotary wing design is today nearly as much an art as it is a science,... that the helicopter has not yet been fully invented."[12] Of course, such statements are exaggerated but they are not altogether incorrect. They reflect the complexity of the rotorcraft and its design. Of course, the helicopter has reached technical maturity, but development today still needs much more time than with fixed-wing aircraft, and the development has been risky, often requiring changes and modifications during flight testing.

The ability of the helicopter for hover and vertical flight and for operations from restricted areas found military interest from the beginning of practical helicopter flight during World War II. Military operations of helicopters had their break-through during the Korean War, and since that time military operations without helicopters would be unthinkable.

In the past, the helicopter has succeeded as an operational vehicle because no other aircraft, no matter how simple or inexpensive, has been able to compete with it in the performance of the particular tasks associated with vertical flight and efficient hover. The present utilization of helicopters finds many and varied uses. In the military role helicopters are employed in tasks as light observation, light tactical transport, medium transport, armed escort, antisubmarine warfare, air/sea rescue, vertical replenishment, in-shore replenishment, heavy lift, general utility, antitank and attack missions. Examples of civil uses are: short haul transportation, police patrol, ambulance and rescue missions, aerial surveys, aerial spraying for agriculture, replenishment and support of off-shore stations, aerial crane for construction, and heavy lift. The strong interest in rotorcraft as a continuing means of both military and civilian transport is a clear indication of the realization of its utility and potential. The unique vehicles employing rotary wings which were considered as oddities of the past, have become in the present, utilitarian vehicles with discreet and distinct missions to perform. Although future application appears to be assured, it is essential that a more effective and efficient machine be developed.[11]

Efficiency is imperative to permit long hover times without comprimising mission accomplishment and economy, and it is important that the rotary wing aircraft be capable of controlled hover and vertical flight to a high degree because of the desire for precise placement of externally carried loads and operation in confined terminal areas. In forward flight the cruise efficiency and productivity have to be improved, and speed and range have to be extended for additional missions.

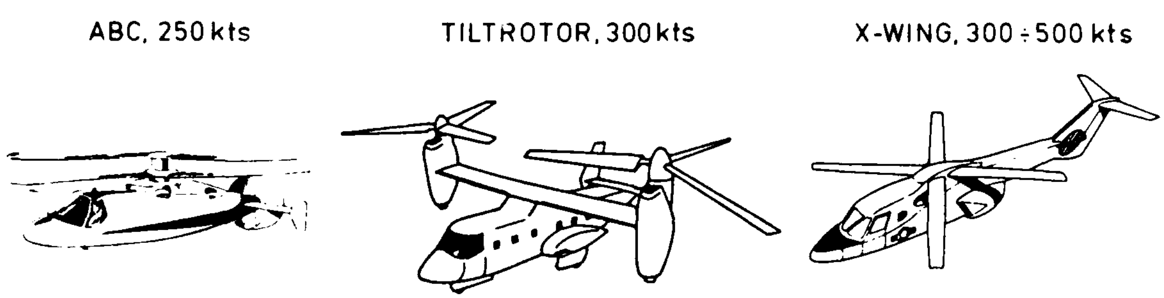

Figure 6 High-Speed Hybrid Rotorcraft

Extention in speed and range will require new concepts. There are advanced rotary wing concepts like tilt-rotor, ABC, X-wing, and compounds (Figure 6), and certain other VTOL technologies, for instance those associated with the moderate by-pass vectored thrust engine, have developed to the point where sucessful aircraft can be realized. Finally, mission requirements and mission effectiveness will decide about the optimal concept. For all missions requiring longer hover phases rotary wing concepts offer best cost efficiency even in future.

1.2 Interaction of Cost and Performance

The use of helicopters in great quantities demonstrates that its mission effectiveness is worth the increase in cost compared to other competing vehicles because it can do something which no other aircraft can do with comparable efficiency. The most important aspects to be considered in defining the cost-effictiveness of military systems are: flight performance, maintenance and reliability, vulnerability, operating costs and logistics. Whereas for civilian application it is not a big problem to describe the utility in a true relation to the expenditure and to show profit or loss, the assessment of utility and cost-effectiveness of military systems is much more difficult and depends upon various assumptions. For military equipment the costs for operation, maintenance and logistics are the most important factors to be examined. This is evident as experience shows that the cost-relation for research and development, procurement and operation is in the range of 1:3:6. This means that it would be worthwhile to spend more money for research and development if the costs for procurement and operation and maintenance could be reduced.

Modern technical systems must frequently meet different requirements; bringing the systems, either in components, or as a whole, to the limits of technical realization. This is especially true when new developments must meet stronger requirements, for instance higher speed or extended range, compared to the previous vehicles. A simplified consideration of costs in relation to technology shows for a certain aim or level in research, development or performance a progressive increase of costs. In the upper region a point of diminishing return of costs, that means a loss of cost-efficiency, will be reached. Figure 7 explains the level of technology as a parameter for costs and requirements. For a given requirement, the state-of-the-art defines the point 0 with a certain amount of costs. Improvement in the technology level, for instance in aeromechanics, in materials or design, will result in a cost reduction or would allow increased requirements at the same cost level. This situation leads normally to the socalled "next generation" product, and the history of aeronautics demonstrates such technology steps by "new technology", for instance the jet engine and the swept wing for fixed-wing aircraft or the transonic blade section and the introduction of composite structures to helicopters[15].

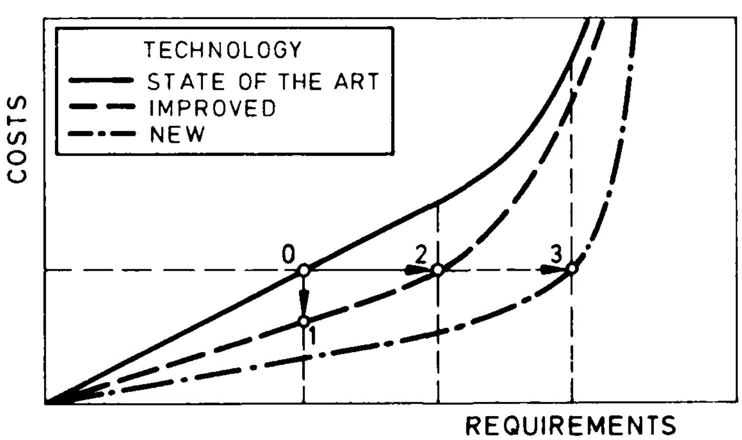

Figure 7 Influence of Technology State on Costs

Invention is an important part of technological progress, which is too often ignored in planning development activities. Future planning should include the promotion and fostering of invention and innovation, and research and development efforts should be structured to take advantage of these aspects when they occur[12]. New or improved products need new ideas or innovations introducing something to bring them in an evolutionary process step-by-step to a better state. Sudden complete or fundamental changes in a revolutionary sense are very seldom in engineering. But no doubt, it is not sufficient for advanced systems to be based just on experience.

The main driving factors for rotary wing technology are hover efficiency, cruise efficiency, speed, range and payload, which are mainly influenced by aeromechanics. In addition, if the rotary wing is to realize its full application, it must be made more reliable and maintainable and brought to the reliability and maintainability levels of fixed-wing configurations. For long time the high mechanical complexity of the helicopter and the resulting high maintenance costs were major disadvantages. By a strong effort the structural situation could be improved with enlarging the lifetime of critical components. Figure 8 illustrates the very strong influence of the Mean-Time-Between-Removal (MTBR) of critical components on the direct operating costs per flight hour. For the older helicopters with short component lifetime, maintenance and parts costs were dominating and the aerodynamic efficiency resulting in the petrol costs was of minor importance to the overall costs. This situation has changed by the reduction of the mechanical costs. Fuel costs get more and more important, especially considering the increase of fuel prices.

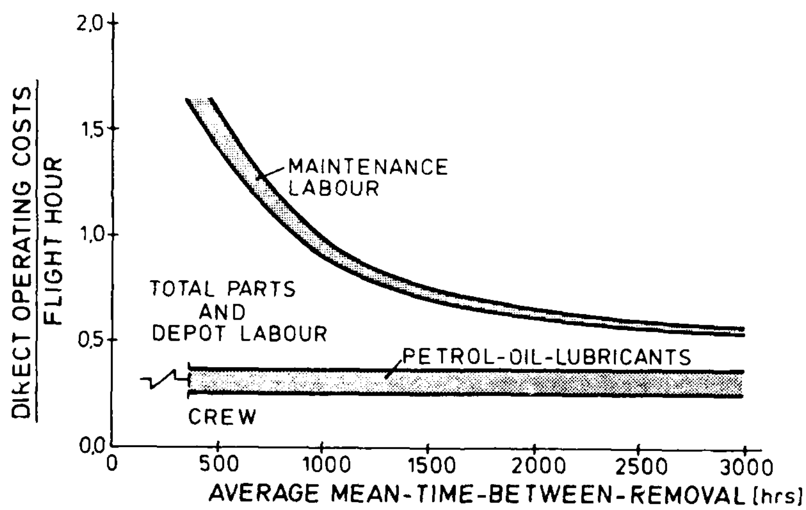

Figure 8 Influence of Improved Reliability and Maintainability on Costs

As experience has shown, new technology very seldom will result in less expensive new products for military applications, because there is always a wish to extend the requirements, and have additional missions with the successor system. This trend is well underlined by the UTTAS (Utility Tactical Transport Aircraft System) and the LHX (Light Helicopter, Experimental) programmes. Figure 9 shows the startling trend in unit cost growth of US Army helicopters. The inevitable consequence of this will be a diminishing inventory of helicopters, albeit comprised of individually superbly capable vehicles. The time would seem to be near at hand when requirements which demand "that last few percent" of performance may have to be foregone, and when advanced technology must be applied to increase numbers through reduced total ownership cost rather than to search for greater performance[12].

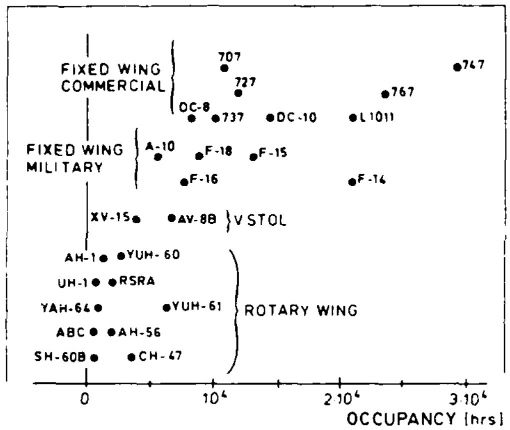

Figure 36 Comparison of Wind Tunnel Test Time

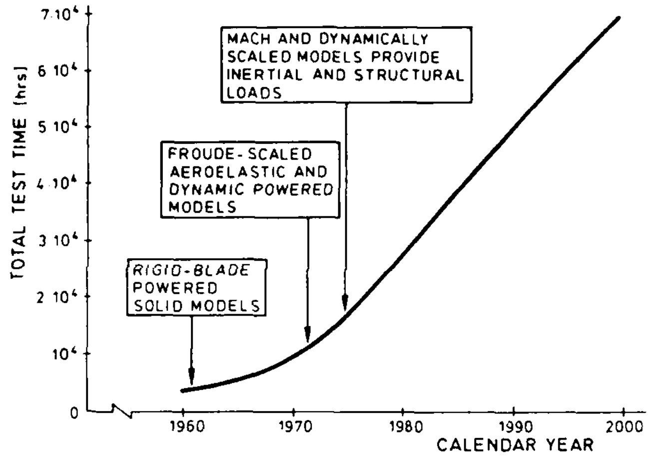

Figure 37 Wind Tunnel Test Hours for Research and Development

Although there is certainly no fundamental law which demands that rotary wing and fixed-wing aircraft engineers adhere to the same design practices and policies, or that they devote comparable resources to corresponding tasks, the disparity in the two fields of endeavor in these regards is nonetheless striking. The traditional practice in the development of helicopters has been to minimize costs associated with the design process and, implicitly, to accept the consequences of unforeseen problems which tend to arise during test flight. The reasons for this approach undoubtedly stem mostly from a reluctance or inability to make enough research and development funding available early in development and to a lesser degree from the fact that analytical and experimental techniques associated with rotary wing aircraft have not yet reached maturity[12]. The development funding for helicopters has always been much lower than for fixed-wing aircrafts in spite of the much more complex aeromechanical and mechanical problems of helicopters.

The costs of correcting basic defects during the flight test phase sometimes reach a similar order of magnitude as the basic development fundings. If the problems had been identified in an earlier step of the development process, they could have been corrected at much less cost in terms of both time and money. Only comparatively small amounts of funds would be required to make significant improvements early-on in a development programme. The risk of development can only be diminished by a combination of more and improved theoretical studies and calculations, by the use of today's possibilities of simulation, and by much more and better wind tunnel work, as indicated in Figure 38.

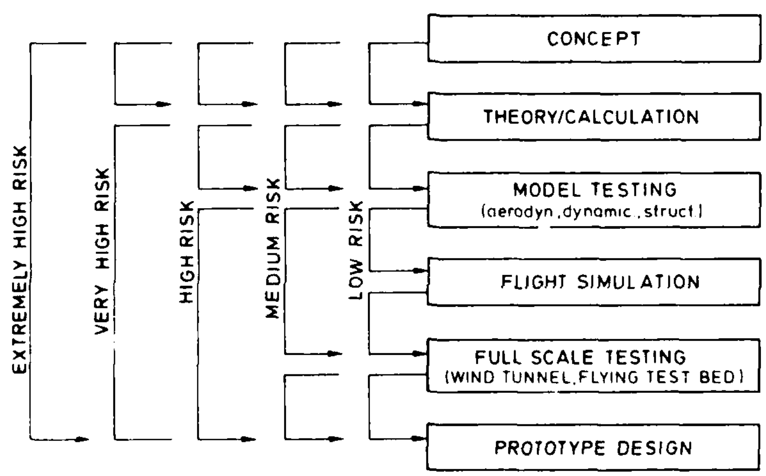

Figure 38 Methods to Reduce Risk of Development

helicopter blade in three dimensions an extremely difficult problem. Again, a large increase in computing capability is required to attack such design optimiziation problems.

In the past, the helicopter industry has been less aggresive in pursuing the potential benefits of large computers than the fixed-wing aircraft community, even though the aerodynamic and structural dynamic phenomena are more complex for rotorcraft. Clearly, the rotary wing community does not consider itself as having a driving need for a supercomputer although such a capability would undoubtedly be used if it became available. There is no doubt that the applications of super computers must generate sufficient confidence in more realistic modelling to pay off in reduced proving trials on the product. This attitude will continue to be manifested in an apparent reluctance by the user community to embrace new theoretical methods until cost effetive applications have been proved.

As an example Figure 34 illustrates the development of the prediction accuracy for the external loads and the goals for future progress. Whereas the prediction of transient loads will remain unsatisfactory for a long time, for steady state loads sufficient accuracy is foreseeable. Absolute accuracy is not possible and not necessary, but it should reach the same order as possible for pure mechanical systems, for instance the weight estimation as shown in Figure 35.

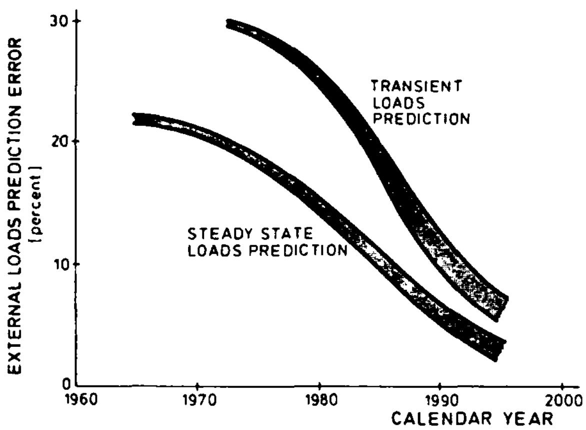

Figure 34 Improvement of External Loads Prediction

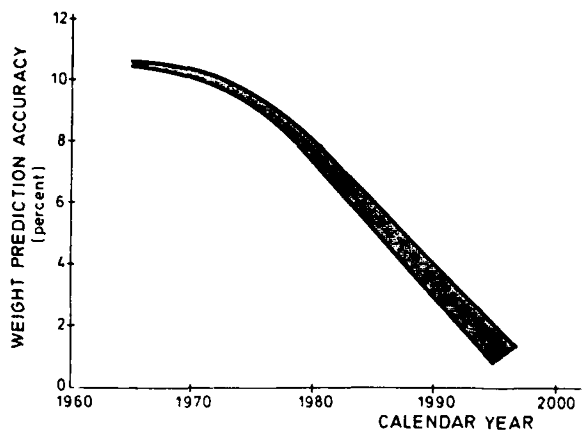

Figure 35 Improvement of Weight Prediction.

With the insufficient theoretical prediction methods it would seem quite natural to extensively use experimental results from the wind tunnel during the development, however, use of the wind tunnel as an important design tool is modest as indicated in Figure 36 by the comparison of wind tunnel test time for fixed-wing aircraft and for helicopters. It is particularly significant that commercial fixed-wing practice relies heavily on wind tunnel data. The reasons for the modest usage for helicopters have to be seen in the model complexity, in wind tunnel wall effects and in the inadequacy of current sub-scale techniques; techniques which must be further developed. Today's possibilities allow study of special phenomena in the wind tunnel in the sense of research, but wind tunnel based predictions for development are as insufficient as those derived from calculation. There is a pronounced trend of an increased use of the wind tunnel as shown in Figure 37, but the future has to prove its qualification as a design tool in order to save time and costs of development.

shown in papers gives a wrong impression in most cases because the theoretical values are not really predicted but result from calculations done after the tests with modified input parameters.

This unsatisfactory situation is the main reason for the long development time of helicopters compared to fixed-wing aircraft as shown in Figure 33 with the time between first flight and certification. This time is normally less than one year for the fixed-wing aircraft but between two and four years for helicopters, indicating that the helicopter needs modifications during the flight test phase because of problems which had not been recognised earlier.

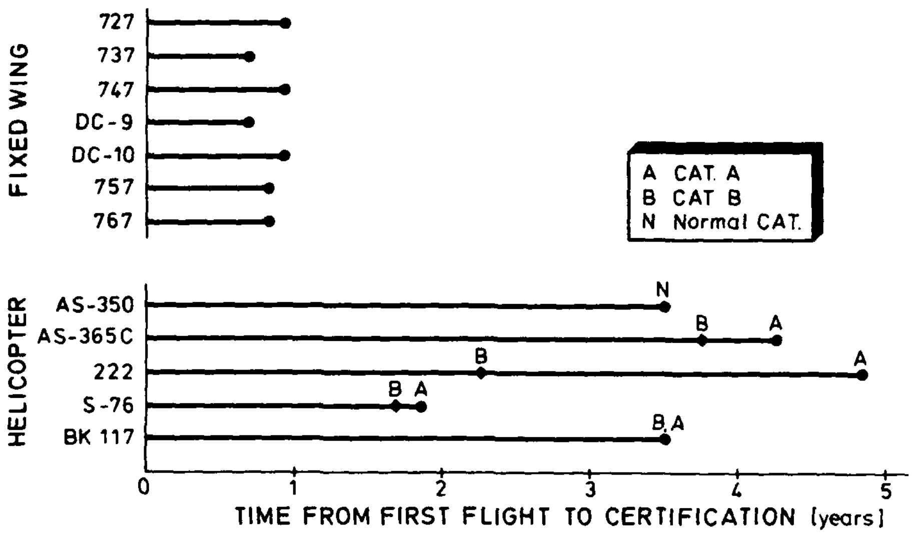

Figure 33 Time for Certification of Helicopters and Fixed-Wing Aircraft

The new generation of computers allowing large scale computing will have potential to improve the situation. In an AGARD Working Group future possibilities have been studied [17]. The so-called free-wake analysis, finite-difference transonic calculations in the lifting line or lifting surface representation of rotor blades, viscous effects in the forms of retreating-blade dynamic stall and advancing-blade shock wave-boundary layer interaction should become possible, but the calculation of the complete unsteady flow field for typical helicopter configurations with the aerodynamic interference between various rotorcraft components is expected to remain beyond the state-of-the-art for the forseeable future.

Large scale computing should also be helpful for flight simulation. Rotorcraft simulation appears to be particularly demanding on computer capability: blade flexibility, radial and azimuthal variations of blade forces and moments, nonlinear effects such as stall or transonic tip aerodynamics, rotor-fuselage flow interference and dynamic interactions all combine to defeat the ability of the largest available computers to provide real time simulation with complete fidelity.

For the aeroelastic and dynamic response of rotary wing aircraft, the computer requirements are substantially greater compared to fixed-wing aircraft requirements because of the gyroscopic coupling effects which introduce structural nonlinearities that add to the aerodynamic nonlinearities. For helicopter rotorblades the situation is further complicated by higher amplitudes of lift fluctuations, a variation of Mach number along the blade due to rotation giving strong transonic effects at the blade tip and the out-of-phase variation of local blade incidence and local Mach number. The inclusion of unsteady nonlinear aerodynamics simultaneously with structural nonlinearities arising from large amplitude deflections make the analysis and computation of the

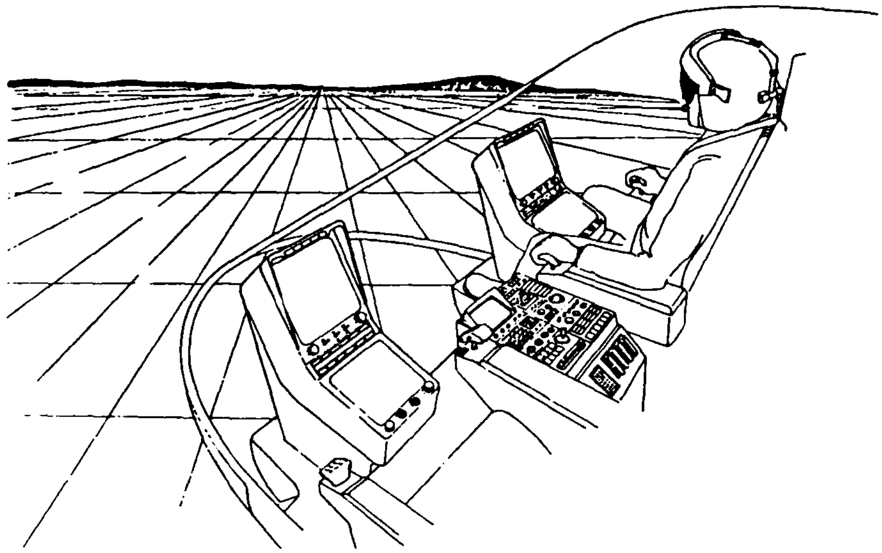

Figure 32 Advanced Helicopter Cockpit Design

2.2 Procedures and Understanding of the Problems

When the first helicopters had been brought to flight, there was practically no real understanding of the physical problems of the helicopter. With the historical development also the knowledge continuously has been established. Around World War II with the first practical helicopters a state had been reached allowing practical calculations and predictions based on theoretical assumptions and theories, which are still today in use. Since that time the procedures have been extended and improved continuously, strongly promoted by the upcoming computer possibilities. Now, there is a relatively good understanding of all typical phenomena. This improved knowledge has finally resulted in the progress of the helicopter.

Nevertheless the state of the available procedures has to be considered to be all but good. The methods are unsatisfactory in the sense of design tools requiring accurate predictions. Concerning aeromechanics it has to be stated that current analytical procedures are limited primarily because of the incompletely defined nature of the unsteady flow fields of rotors in the presence of fuselages through a very wide angle-of-attack range, and the extremely complex, usually non-linear, multi-frequency structural dynamic interactions. Performance,stability and control predictions appear to be of acceptable accuracy except for operating conditions at high advance ratios and thrust coefficients and the higher angle-of-attack ranges. Nap-of-the-earth handling qualities are predictable except for the interaction effects of the fuselage and main and tail rotors at low flight speeds close to the ground. Unfortunately these areas are important parts of the operating range. Reliable methods for fuselage vibration and oscillatory structural loading predictions, especially for the higher frequencies, are not available. Small changes to existing components may cause significant changes in the vibration level which are not readily predictable.

With the currently available computers it is not possible to use rather sophisticated theories because of their extreme complexity. On the other side such theories would be too difficult in handling for engineering purposes. Therefore, engineering approaches are widely used with limited accuracy, normally sufficient for preliminary design, but keeping the risk of development until flight test phase. The perfect correlation often

Surveying the trend curves of helicopter technology would be incomplete without the weight breakdown trend. The reduction of the empty weight is mainly due to structural improvements and lower weight of the different aircraft systems. The aircraft weight fraction improvement trends are shown in Figure 30. The all-composite aircraft offers additional potential.

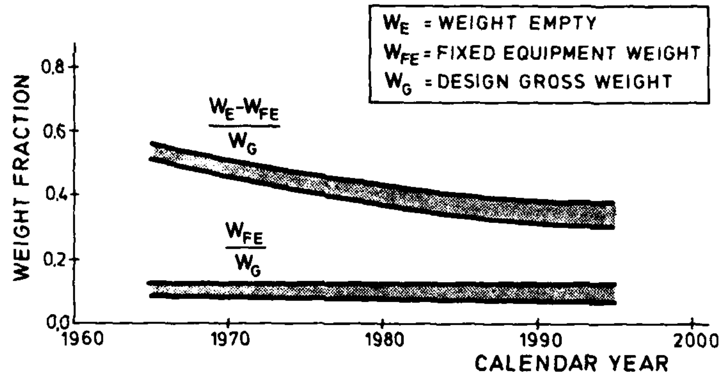

Figure 30 Helicopter Weight Fraction Improvement Trends

Flight control work has resulted in many benefits which include reduced pilot work load and increased rotorcraft safety, and performance. Active controls technology has been considered for reducing rotor dynamic loads and aircraft vibration. Industry is now moving into fly-by-wire control and eventually will employ fly-by-light. Figure 31 illustrates the historical development of helicopter flight controls. The possibilities of modern electronics, avionics and optronics will have a great impact on cockpit design and pilot's workload as indicated in an advanced helicopter cockpit design in Figure 32, but will influence all helicopter systems as well.

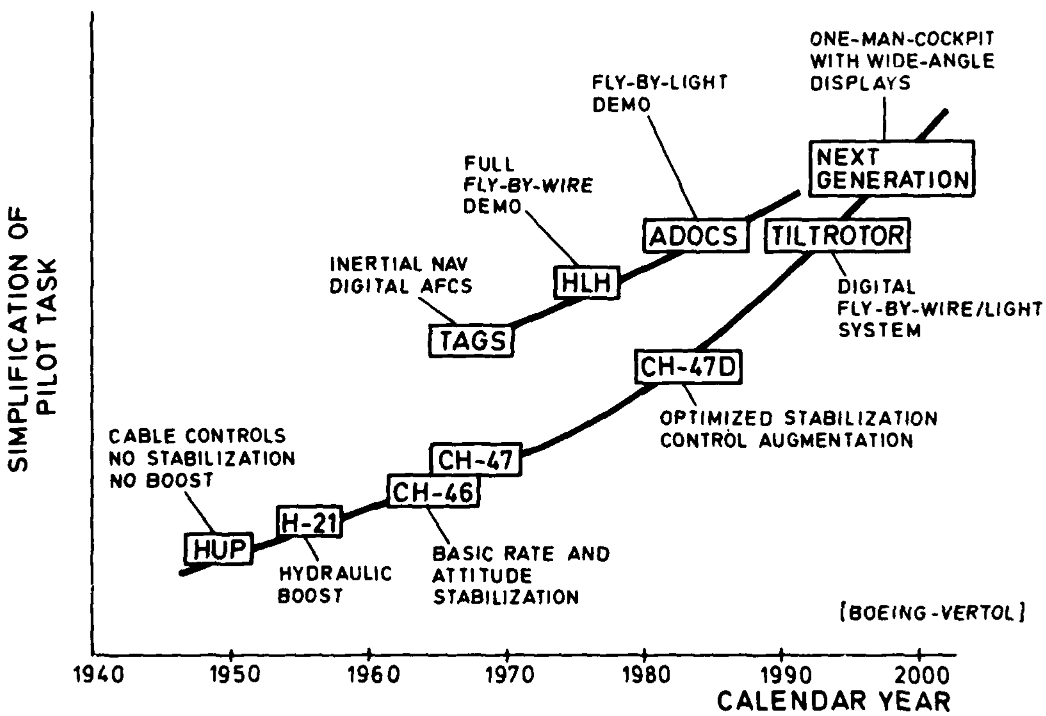

Figure 31 Historical Development of Helicopter Controls

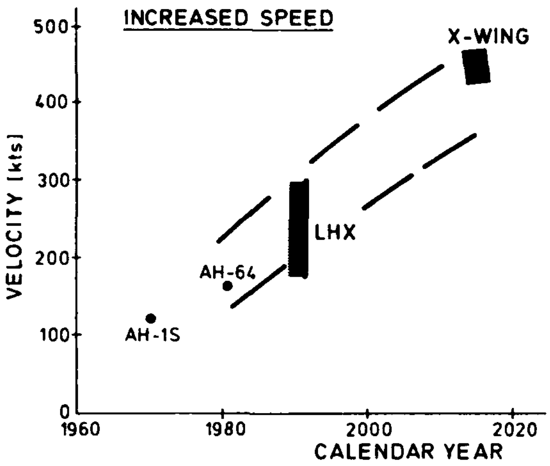

Figure 25 Trend of Speed Increase

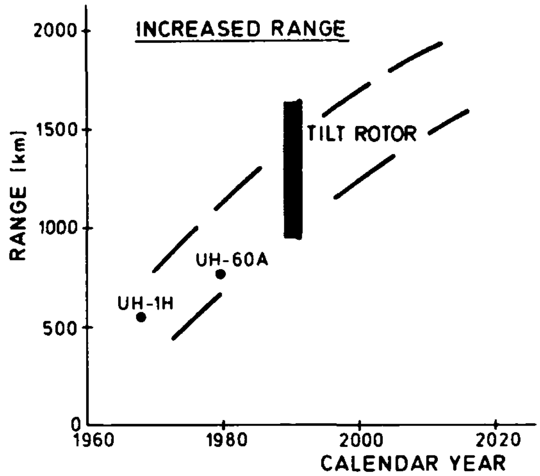

Figure 26 Trend of Range Extension

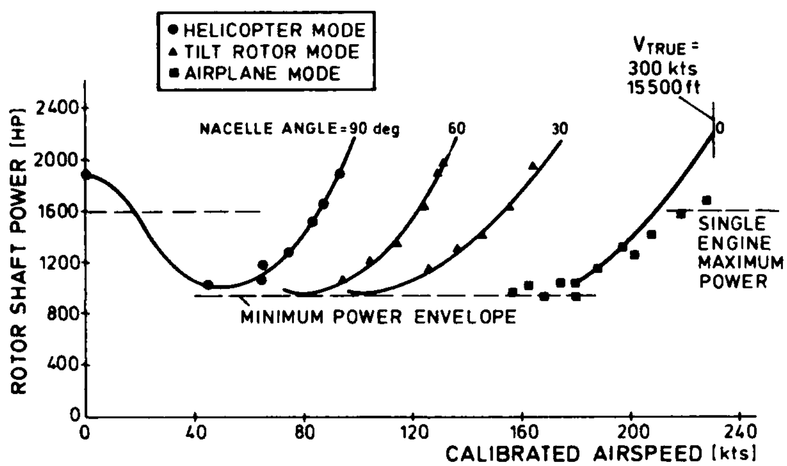

Figure 27 XV-15 Required Power for Level Flight

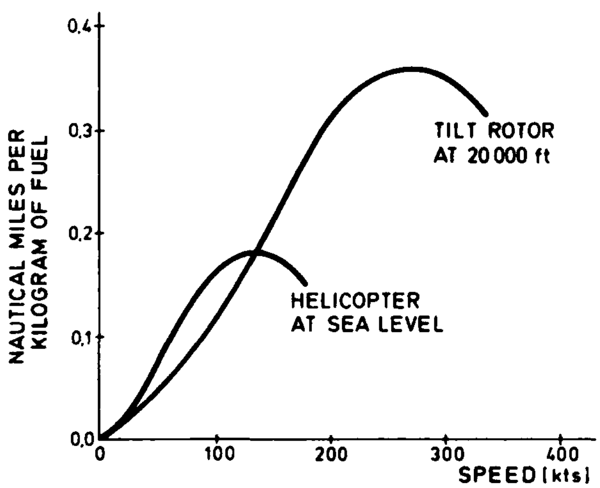

Figure 28 Fuel Efficiency Comparison

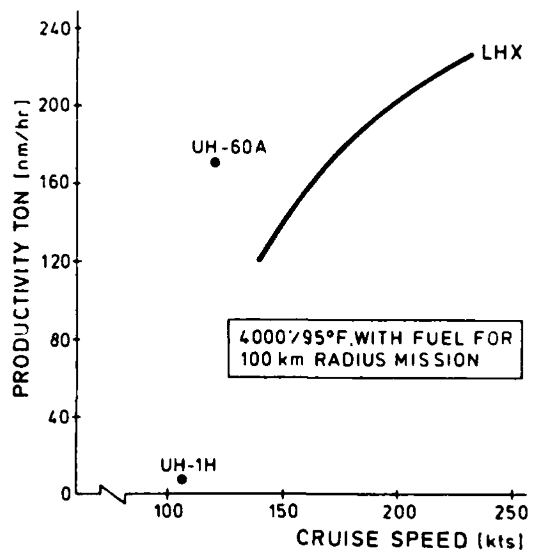

Figure 29 Utility Aircraft Productivity

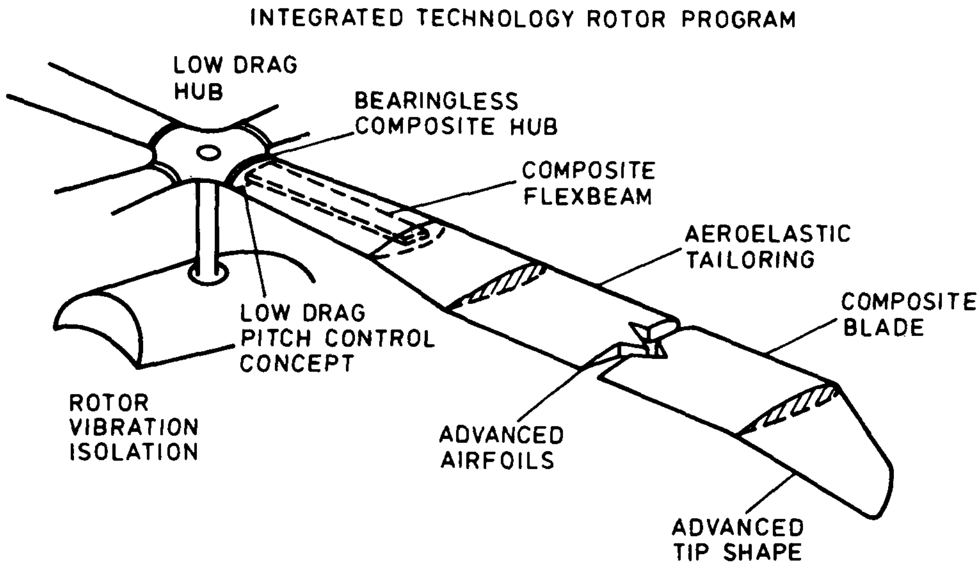

Figure 23 Advanced Bearingless Rotor Concept

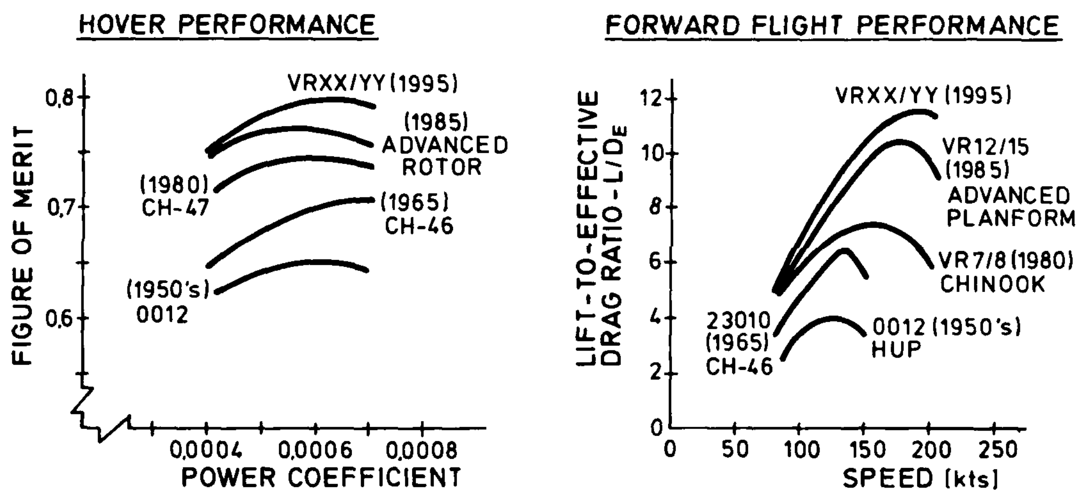

Figure 24 Aerodynamic Rotor Efficiency

efficiency is characterized by figure of merit and lift-to-drag ratio as shown in Figure 24. It is worthwile to note that many current systems fail in the utilization of their full performance inherent in the design as a result of unsatisfactory operating characteristics or dynamical limitations with high vibrational loads. In such cases, there is great opportunity and potential for improving the operating charcteristics to a degree that the full performance potential inherent in the design may be realized.

The extension of speed and range with more advanced concepts like compound, tilt-rotor and X-wing is illustrated in Figure 25 and Figure 26. The tilt-rotor aircraft can vary flight speeds over a wide range at minimal power settings. This characteristic is shown in Figure 27 by the measured power required by the XV15 to maintain level flight at different conversion angles of the rotors. The fuel efficiency of a helicopter and a tilt-rotor aircraft is compared in Figure 28, and Figure 29 indicates the increase of productivity by a comparison of helicopters from different generations.

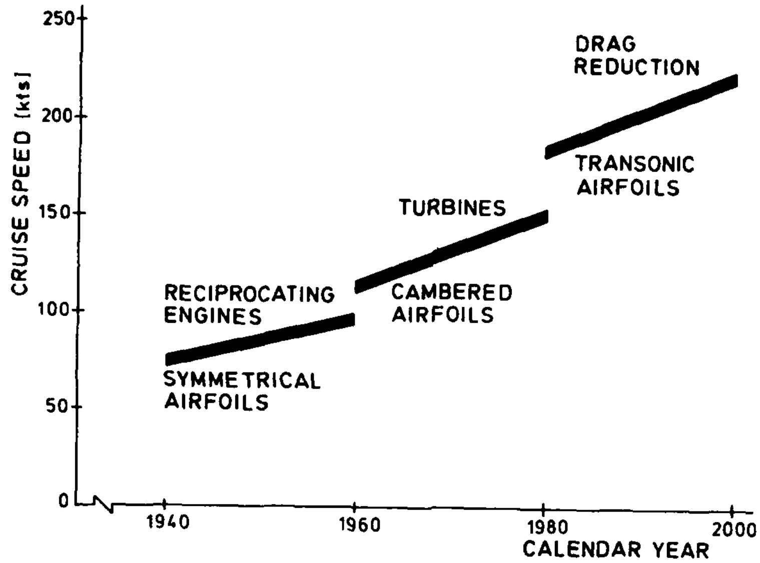

Figure 20 Speed Increase by Advanced Technology

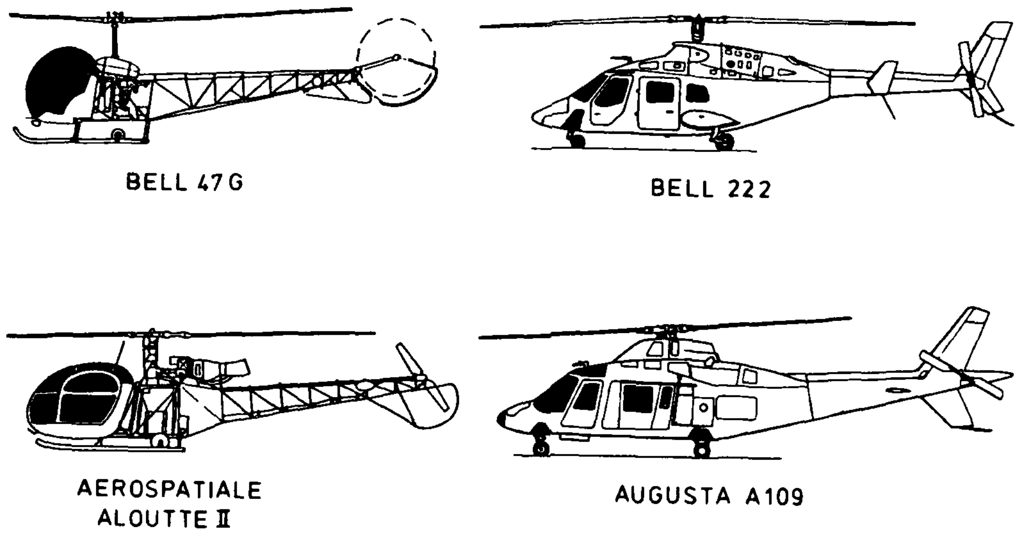

Figure 21 Aerodynamic Shaping of Helicopters

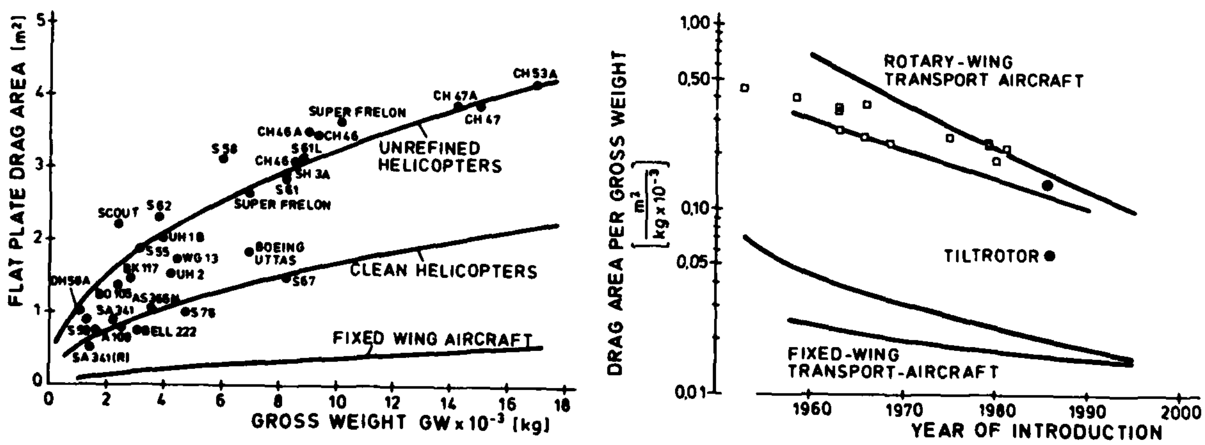

Figure 22 Airframe Drag Trend

GENERATION	1ST	2ND	3RD	4TH
DESIGN PHILOSOPHY	SAFE LIFE →		← FAIL SAFE →	
	Symmetrical Airfoils	Cambered Airfoils	Peaky Airfoils	Integrated Rotor
Aeromechanics	Rotor L/D = 6.8	7.3	8.5	10.5
	Helicopter L/D=4,1	4.8	5.4	6.6
Propulsion	Reciprocating	1st Generation Turbine	2nd Generation Turbine	3rd Generation Turbine
	Specific Wt=1.2SHP/lb	2.2	2.6	3.6
	SFC=0,54 lb/SHP/hr	0.65	0.59	0.46
	Specific Vol=9.4SHP/ft^3	11.2	13.1	16.6
Structures	AluminumSkinandStringer	BondedPanel,SecondaryComposites		PrimaryComposites,
	Simple Crew Restraint	Crashworthy Fuel System		CrashworthyFuelSystem
	UsefulLoadFraction=0.65	0.46	0.43	0.37
Avionics	Vacuum Tube Comm	MilitaryStandardComm	VLSIC Comm/Nav	Integrated Cockpit
	Commerical Nav	Initial Solid State	IntegratedNight	2nd Gen Night
	Austere Instruments	IMC Instruments	Vision, Self-	Data Bus
		Commerical Nav	Contained Nav	
			Discrete Info	
			Channels	

1950 1960 1970 1980 1990 2000

Figure 19 Helicopter Technology Evolution

all. For example, rotary wing aircraft regularly operate in unsteady flowfields, in ground effect, with near-sonic flow conditions on lifting surfaces and in a condition of static instability. There are complex and significant interactions between the lifting rotor and other components of the helicopter, particular at low speeds, which are just beginning to be understood. Small and seemingly unimportant design changes can provide large changes in the vibration and stress levels encountered in flight as a result of the interactions. These effects all combine to make the development of rotary wing aircraft an extremely complex undertaking which should be initiated only with the due recognition of this fact and with the willingness to devote appropriate resources in order to control the inherent risks.[12]

Technology advancements have been critical to the succes of rotorcraft applications specifically in the areas of aeromechanics (aerodynamics, dynamics, aeroelasticity, flight mechanics and controls, acoustics), propulsion (power plants and transmission), structures (materials, design, crashworthiness), aviation electronics and the integration of these technologies into the entire rotorcraft system. The different papers of this Lecture Series, however, address in detail only the role of aeromechanics.

The advance in helicopter technology is illustrated by the following diagrams. Figure 20 shows the speed increase of helicopters since the 1940s resulting from improved rotor aerodynamics, from drag reduction and higher installed power. The aerodynamic drag situation has been greatly improved with today's helicopters compared to the older helicopters illustrated in Figure 21, however, compared to fixed-wing standards the drag level of helicopters is still much too high with great potential for further improvement - Figure 22. The aerodynamic efficiency of the rotor is primarily influenced by the airfoil, the twist and geometric shape of the blade. Aerodynamics and dynamics of the rotor have to be considered with their interaction and in connection with the overall characteristics influenced by the design as illustrated in Figure 23. Aerodynamic rotor

Completion and flight testing of the US Army/Boeing Vertol XCH-62 Heavy Lift Research Vehicle (HLRV) will lay the groundwork for an Army technical data base leading to an advanced cargo rotorcraft (ACR). The XCH-62 (Figure 18) will lift about 35 tons. Current U.S. cargo and transport helicopters have a maximum 16-ton payload capacity. First flight is expected in 1988. This programme is a continuation of the HLH programme which had been interrupted in 1975 for budgetary reasons. The XCH-62 programme will resolve operational issues that will have to be addressed before proceeding with the ACR. The ACR may be either a helicopter or a tilt-rotor aircraft, but dynamics, gears, transmissions, control systems and some other aircraft systems will be assessed and developed with the HLRV[16].

The next generation helicopter will have higher speed, heavier lift, and longer range as natural means of improving productivity, while fundamental improvements in reliability, vibration, noise, and safety are necessary elements of increasing cost-effectiveness. Advances such as improved rotors, reduced drag, light weight composite structures, and more efficient engines will have a great impact on the new generation helicopters.

Much of the technology developed to meet the military requirements will be directly applicable to future designs for commercial operators, who are expected to demand lower cost, light weight, high speed and efficient vertical takeoff aircraft that have day/night, allweather IFR capabilities. Also commercial applications for tilt-rotor transports derived from the JVX are expected, such as passenger city-center VTOL transports and as high speed supply aircraft to serve large oil rigs.

2. THE DEVELOPMENT OF HELICOPTER TECHNOLOGY

2.1 Technological Opportunities - Past, Present, Future

The appearance of the helicopter did not change drastically from the first practical helicopters of the 1940s up to today's helicopters compared to the rapid change of generations of fixed-wing aircraft during the same period of time. It is useful to recall that helicopters are only now in their third generation. The following table (Figure 19) gives a survey of the helicopter technology evolution showing that there have been not too many significant changes. The only abrupt change was in the propulsion system with the introduction of the shaft turbine which replaced the reciprocating engine. All the other changes have been gradual - the aerodynamic improvement of the rotor by better airfoils, the reduction in parasite drag to achieve a better L/D, the structural changes with the introduction of composites, and the changes in electronics/avionics. Of course, in detail, all components and systems have gone through many modifications and changes, and have a much better performance and reliability today.

Helicopters have found increasingly greater applications in fulfilling both military and civil requirements of air mobility. Because of the permanent interaction between requirements and activities for technical improvement, the technical possibilities have changed, a process in which competition between the manufacturers was greatly helpful. Although the current generation helicopters have greatly improved from the previous generations, rotorcraft technology, in a technical sense, is still immature, and potential improvements are envisioned.

The reason for the slow progress in rotary wing technology over the years has to be viewed in the complexity and deficient understanding of the problems. The aeromechanical and structural complexities of a rotary wing vehicle are in fact enormous. Rotary wing vehicles regularly operate in difficult portions of the flight envelope which fixed wing aircraft tend to encounter only on a transient basis or not at

Figure 15 German PAH-2 Model

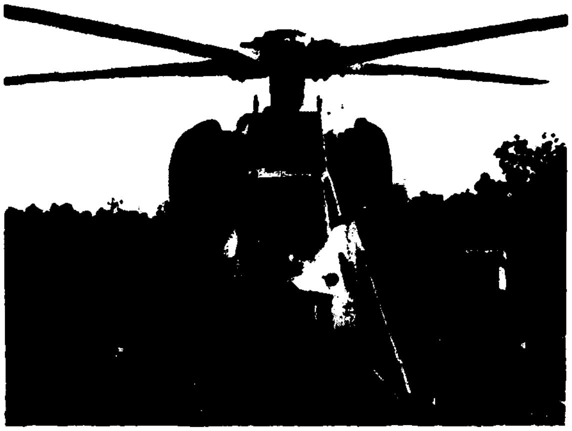

Figure 16 The A 129 Mongoose Light Attack/Scout Helicopter

Italy is accelerating the flight programme of its armed helicopter, the Agusta A 129 Mongoose/Mangusta (Figure 16). Design freeze will be by the end of 1985. The A 129 is an twin engine aircraft in the 3700 kg weight class with an advanced electronics/avionics/ optronics system.

Britain has yet to decide whether to participate in one of these programmes or to develop its own armed helicopter based on a Lynx version.

Nato has a study programme going for the NH-90 helicopter involving France, Germany, Italy, Holland and Britain. The programme is oriented to a battlefield transport helicopter in the 6 - 8 ton class. The NH-90 project also is planned to be produced in a naval version, and several countries will be looking for replacements for their current shipborne helicopters in the 1990s.

Italy and Britain both have requirements for a large antisubmarine (ASW) helicopter to replace the Sikorsky and Boeing designs the two navies use. The EH-101 will be in the 30,000 lb. maximum takeoff weight category.

Figure 17 A Modell of the Westland-Agusta EH-101

Figure 18 Boeing Vertol Heavy Lift Helicopter HLH

ROLES/MISSION

COUNTRY	SCOUT	ATTACK	UTILITY	TRANSPORT	MED/HVY LIFT	ASW/SAR
United States	LHX	LHX	LHX			
				JVX	JVX	JVX
					ACRV, HLH	
Germany-France	PAH-2	PAH-2				
England-Italy				EH-101		EH-101
England	LYNX-3	LYNX-3				
Europe				NH 90		NH 90

Figure 13 Advanced Rotorcraft Programmes

appears to be no consensus on the speed issue, and some see the requirement for a nap-of-the-earth operating capability on one hand and high speed on the other to be basically contradictory. Current Army planning envisions the procurement of about 4,500 LHX aircraft in both the scout/attack and utility configurations[16].

The U. S. JVX programme, which will result in a tilt-rotor assault transport tailored for use by the Navy, Marine Corps, Air Force and, in the future, the Army also represents a significant technical and systems integration challenge. The JVX is being developed by an industry team headed by Bell and Boeing Vertol. The JVX design draws on experience gained in the Army/NASA/Bell XV-15 experimental tilt-rotor programme. Figure 14 illustrates the significant features of the JVX tilt-rotor configuration. It will be powered by two 5,000-shp. engines driving 38-ft.-dia., three bladed composite rotors. The aircraft will carry 24 troops at a cruise speed of 250 kt. The tactical range is specified to 1,400 naut. mi., and the ferry capability to 2,100 naut. mi..[16]

Figure 14 JVX Tilt-Rotor Configuration

The European helicopter manufactures are forming multinational groups for the new programmes. The armed helicopter programme is receiving the most attention at present, as several European nations have pressing requirements. Development of the Franco-German PAH2/HAP/HAC helicopter is being done jointly by MBB und Aerospatiale. There will be three basic versions of the PAH-2. The German PHA-2 will be primarily an antitank helicopter, but this version can also carry missiles to be used for short-range air to-air defense. The French versions are an antitank helicopter (designated HAC) and an escort aircraft (designated HAP). The maximum takeoff weight in the 10,000 - to 11,000-lb. range will be about half that of the AH64. Figure 15 shows a model of the German version.

A. SCOUT/ATTACK

MANUFACTURER	SCOUT	SCOUT/ATTACK	ATTACK
Bell	OH-13 (Sioux) (Modell 47) OH-58A,C,D (KIOWA/AHIP) (Modell 206 Jet Ranger)		AH-1G, J, S (Cobra)
Hiller	OH-23 (Raven)		
Hughes	OH-6A (Cayuse)	500 MD (Defender)	AH-64 (Apache)
Aerospatiale	SE 313B/SA 316B (ALOUETTE II/III)	SA 341/SA 342(Gazelle)	
Agusta		A 109A/A129 (MONGOOSE)	
MBB	BO 105	BO 105	BO 105
Westland	WASP/SCOUT	AH-1(LYNX)	

B. TRANSPORT/UTILITY (<30,000 lbs)

MANUFACTURER	TRANSPORT/UTILITY	ARMED	AEW/SAR
Bell	UH-1 D, H, N(Iroquois) 412	UH-1 B, C	214ST
Sikorsky	UH-60A (Black Hawk) CH-53	UH-60	SH-3D(SeaKing),HH-3F (Pelican) Lamps MK III
Boeing	CH-46 (Sea Knight)	CH-46	HH-60D (Night Hawk) EH-60A
Aerospatiale	SA 365 F (Dauphin 2) SA 330 L (Puma) SA 332 B (Super Puma) SA 321 G (Super Frelon)	ALL	ALL HH-65
Agusta	A109	A109	
Westland	WESSEX HU.2/HU.5/HAS.3	Wessex	Wessex Lynx

C. MEDIUM/HEAVY LIFT

MANUFACTURER	MEDIUM (30,000 to 60,000 lbs)	HEAVY LIFT (60,000 lbs)
Boeing	CH-47 C,D (Chinook)	
Sikorsky	CH-54A (Tarhe)	CH-53A,D,E (Super Stallion)

Figure 12 Today's Military Helicopters (Western World)

U.S. helicopter manufacturers expect the LHX (Light Helicopter) programme to be the toughest challenge they have faced to date. The Army is scheduled to issue the required operational capability (ROC) in March '85, and the service is expected to specify an 8,000 - 8,500-lb. machine that can be built in scout/attack and utility versions with a high degree of commonality. The Army wants the scout/attack version designed as a single-pilot helicopter. This means significant advances will have to be made in automated controls to reduce the pilot's workload while he is performing the scout mission or fighting air-to-ground or air-to-air engagements. The airspeed requirement could determine whether the LHX will be based on conventional rotorcraft concepts or will have to rely on advanced concepts such as tilt-rotor or compound rotor/wing designs. A cruise speed above 200 kt would exclude most conventional rotorcraft technology from the competition and necessitate alternative solutions such as the tilt-rotor design used by Bell and Boeing Vertol in the JVX programme or the counterrotating main rotor system used in the Sikorsky advancing blade concept (ABC) helicopter (see Figure 6). But there

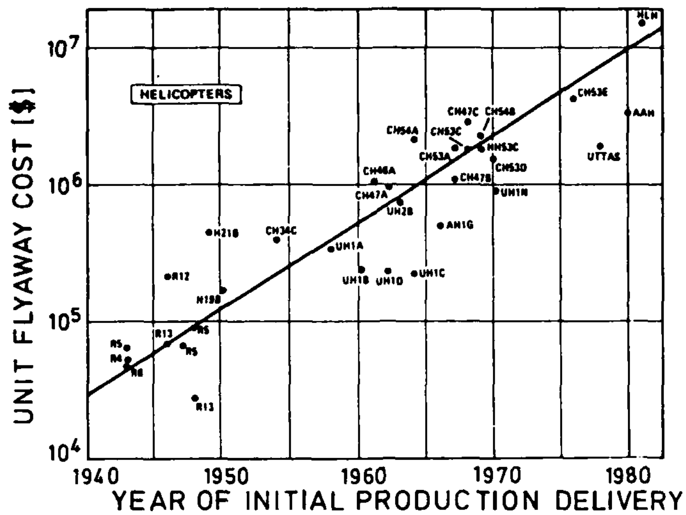

Figure 9 Unit Cost Increase with Time

1.3 Review of Main Helicopter Programmes

The following table (Figure 12) gives a survey of today's military helicopters in the western world, arranged as to manufactures. For those familiar with the names and military designations, it is quite obvious that rather old helicopters are still in military operation, which must be replaced in near future.

The major activities of the past decade were the development of the UTTAS (Utility Tactical Transport Aircraft System) with the UH60A Blackhawk (Figure 10) and the AAH (Advanced Attack Helicopter) with the AH-64 Apache (Figure 11). In addition, several improvement or modernization programmes and adaptions to military missions have been completed.

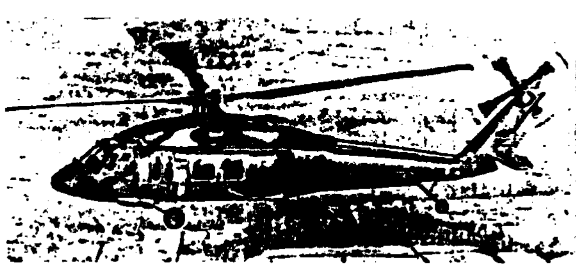

Figure 10 UTTAS-UH-60A Blackhawk (Sikorsky)

Figure 11 AAH-AH-64 Apache (Hughes)

UTTAS and AAH reflect today's helicopter technology in their class of about 8 to 9 tons, whereas the modern technology of light helicopters is pronounced in several helicopters developed mainly for the civilian market, i.e. Bell 222, Sikorsky S 76, Aerospatiale SA 360/365, Agusta A 109, MBB-KHI Bk 117.

The military market needs a new light helicopter, which cannot be based on the existing new civilian helicopters because nowadays military and civilian missions and requirements are quite different. Extended and new missions will necessitate additional development programmes. The following table (Figure 13) lists the advanced rotorcraft programmes which already have been started or soon will start. These programmes will result in the next generation of technologically advanced rotorcraft.

3. CONCLUDING REMARKS

The future effectiveness of vertical lift endeavors depends heavily on the priorities established for technology maturation in light of the potential payoffs of each. The payoffs of specific research and development programmes can be rotorcraft with productivity, cost, and dependability levels approaching conventional fixed-wing subsonic aircraft performance. For different categories, the payoffs can be summarized as follows[13]:

Next Generation Helicopter

- 100% productivity improvement
  - 225-250 knot speeds
  - 30% range increase
  - 40% fuel reduction
  - 30% weight fraction reduction
- Lower external and internal noise - community acceptance, passenger and crew acceptance
- All-weater, zero-zero visibility operations
- "Jet smooth" vibration level
  - Passenger and crew comfort
  - Elimination of vibration related failure
  - Longer component life
- Improved safety
  - lower pilot workload
  - fail safe structure
- Reduced operating cost (50%)

Large Helicopter

- 100% greater payload capability
- 5-fold increase in number of passengers

High-Speed Rotorcraft

- 100% to 200% speed and range increase (relative to conventional helicopters)
  - 400-550 knot speeds
  - 600-1000 mile range
  - Reasonable hover and low-speed performance.

There is good potential for decided improvements. Among the various technologies, which apply to rotorcraft, aeromechanics plays a key role, only structures and materials offer a competing potential for improvement. But there is no doubt, that the goals can only be reached after considerable expenditure to research and development efforts. Special consideration should be given to the necessary understanding of the complex rotary wing problems and to the engineering procedures to be used for developments without to high risk.

The effort on the advanced technologies for future rotorcraft must be aggressively pursued now in joint programmes with industry, research laboratories and universities, and international cooperative technology programmes should be extented. International cooperation has always been stimulating. Strong technology transfer restrictions could result in a technical isolation with consequences to future progress.

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A SURVEY OF RECENT DEVELOPMENT IN HELICOPTER AERODYNAMICS

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This paper addresses various aspects of helicopter aerodynamics. It deals with the aerodynamics of isolated rotors and fuselages as well as with some topics in interactional aerodynamics.

After a brief review of conventional methods, we illustrate the progress made in the last decade on a few typical examples. We describe shortly the new theoretical and experimental approaches presently used by helicopter designers and those which are being developed. We limit the scope of the paper to phenomena having a significant influence on helicopter performance. We discuss computational methods of global rotor performance prediction and show the possibilities offered by more elaborate methods which take into account dynamic stall, three-dimensional unsteady transonic flow and fuselage interference effects. The case of the rotor in hover is also outlined. Some typical wind tunnel and flight test results are shown and the problems related to their analysis are discussed. We end this section by rotor design considerations emphasizing airfoil selection criteria and the optimization of blade planform and tip shape.

Helicopter fuselage aerodynamics will probably remain a subject of experimental investigation for years to come because of serious modelling difficulties associated with strong three-dimensional effects and large regions of separated flow to which the presence of the rotor hub adds further complexity. Flight tests are not very well suited to the detailed analysis of the flow situation and wind tunnel experimentation still remains the most appropriate and widely used method of investigation in spite of certain difficulties associated with the testing of small scale models which are recalled in the paper. Experimental data on fuselage drag is then analyzed and the effect of fuselage and rotor hub wake interference with the stabilizers is discussed.
As regards theoretical work, the paper describes the major computational codes developed in the last decade. After a brief description of the different flow conditions encountered over the fuselage, the simple potential flow pannel methods are reviewed, followed by two-dimensional boundary layer methods which are applied along fuselage streamlines to the determination of the flow separation line. We then introduce various wake models based on the vortex sheet singularity concept.
A critical analysis of the results predicted by these methods shows the necessity of using three-dimensional boundary layer calculations and more accurate but also more complex representations of the fuselage wake.

1.0 MAIN ROTOR AERODYNAMICS

The main rotor is the most characteristic component of a conventional helicopter generating the lift, propulsive forces and control moments required in hover and forward flight.
The aerodynamic operating conditions of the rotor in hover and in forward flight are very different, and will thus be considered separately here, keeping in mind that the design of the rotor is a compromise between the requirements of both flight configurations.

In forward flight, each blade section is subjected to: (1), sinusoidal variations of the attack velocity normal to the blade, (2), sinusoidal variations of the aerodynamic sweep angle, and (3), cyclic variations of angle of attack, with smaller angles on the advancing blade and higher angles on the retreating blade.
A typical example of constant Mach, sweep and angle of attack lines is shown in figure 1. As the helicopter airspeed increases, the dissymmetry between the advancing and retreating blades also increases. This phenomenon has several consequences: (1), small or even negative angles of attack at the tip of the advancing blade together with high transonic Mach numbers, (2), very large angles of attack on the retreating blade, with low incident Mach numbers, and (3), intermediate angles of attack and Mach numbers giving high lift and drag on the fore and aft blades which provide most of the lift and propulsive forces.
A number of non-linear unsteady problems thus arise that will be examined later in detail: (1), 3-D transonic flows on the advancing blade, (2), dynamic stall on the retreating blade, and (3), positive and negative sweep angles on the fore and aft blades.

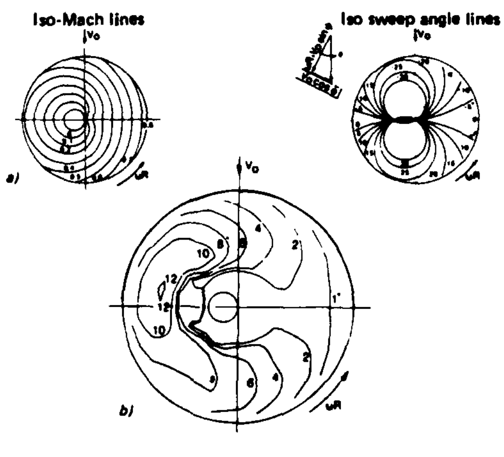

Fig. 1. Typical iso-Mach, iso-sweep and iso-angle of attack lines

1.1 COMPUTATIONAL METHODS

The blade element theory will not be discussed in detail here; a full description can be found, for example, in references [1] and [2]. Briefly, the following parameters are determined for each blade section: (1), the attack velocity conditions, which depend primarily on helicopter airspeed and rotor RPM, and (2), the angle of attack conditions resulting from blade twist, collective and cyclic

pitch control settings, rotor disk incidence, blade motion and deformations, and the induced flow field through the rotor disk.
The general blade deformation equations express the fact that the elastic forces due to blade twist and bending are balanced by the aerodynamic and inertia forces acting on the blades. These equations are defined, for example, in reference [3].

In the simple case of a rigid blade articulated about a flapping hinge the total flapping moment of the aerodynamic and mass forces is zero. The result is a second-order differential equation for the flapping motion (β) in which the right hand term represents the moment of the aerodynamic forces integrated along the blade span:

$$\int 1/2\rho(\Omega r + V\cos\alpha_D \sin\psi)^2 (r - a) Cl(\alpha) dr$$

Assuming small angles, the local angle of attack (α) can be approximated by:

$$\alpha = \theta_0 + \theta_1 \cos\psi + \theta_2 \sin\psi - (v_i + \dot{\beta} r - V\sin\alpha_D)/(\Omega r + V\cos\alpha_D \sin\psi)$$

The aerodynamic characteristics of the airfoils are generally obtained from 2-D wind tunnel test data: Cl, Cd, Cm versus α and M. Corrections may be applied for aerodynamic sweep, unsteady lift and tip relief effects.

The local angle of attack also depends on the induced velocity field (Vi) across the rotor disk which can be computed from the knowledge of the rotor wake geometry or approximated by applying global or local momentum theory.

An equally important parameter is the rotor angle of attack (α_D) which depends on the trim state and more specifically on the helicopter drag/weight ratio.

Finally, the local section angle of attack is also influenced by blade flapping and elastic twist.

Helicopter manufacturers and research groups use various computer codes for calculating the overall rotor performance and loads. In the following paragraphs, these are reviewed and the theoretical predictions compared with experimental results to show their accuracy. We will not discuss here the important question of blade dynamics which is the subject of another conference in this lecture series.

1.1.1 Simple Blade Element Theory

For routine performance calculations AEROSPATIALE uses a program that solves the rigid blade differential flapping equation based on the simple rotor downwash model proposed by DREES [4], which provides reasonably accurate rotor performance predictions, not only for the rotor power requirement, but also for the blade flapping motion producing the rotor pitch and roll moments.

Figure 2 concerns three rotors tested in ONERA's S1 wind tunnel at MODANE, with different airfoil distributions for which 2-D experimental data was available. Rotors 7A and 7B (described in §1.3.1.) were clearly found to have better efficiency than rotor 6B. This trend was correctly predicted by theory although the computed power requirement was in some cases underestimated by as much as 6%.
However, in a comparative assessment of rotors 7A and 7B, which differed only in the airfoil sections between 0.9R and R, the computed results predicted a slight advantage of rotor 7B over rotor 7A at high speeds, whereas in fact rotor 7A was experimentally the better of the two as can be seen on figure 3. Although the discrepancy did not exceed 6 % power, this example clearly demonstrates the limits of a method that uses only 2-D steady airfoil data. The major obstacle concerns the drag coefficient (Cd) which is given as a function of angle of attack and Mach number, since, as discussed in §1.1.6., 3-D unsteady rotor airfoil operating conditions can result in pressure distributions very different from those encountered in 2-D steady flow.

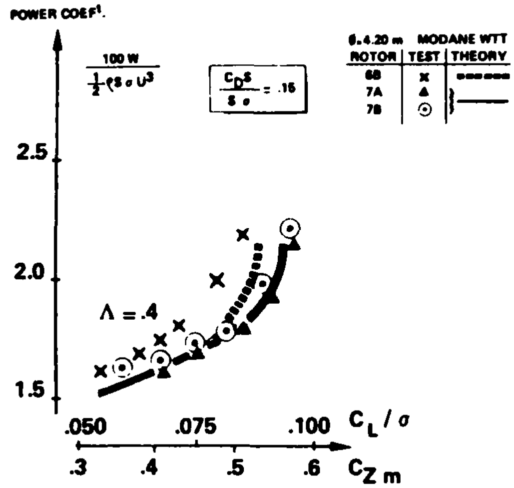

Fig. 2. Test/theory comparison of power vs. blade loading for various AEROSPATIALE rotors at μ=0.4, from [5]

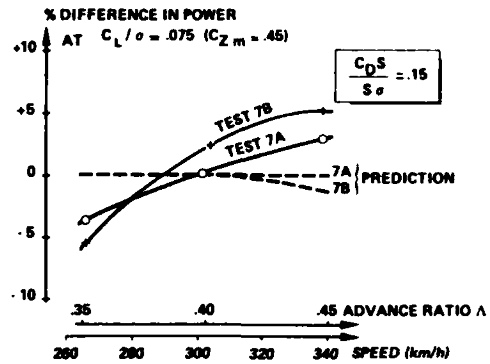

Fig. 3. Power prediction vs. test results on AEROSPATIALE rotors, from [5]

The same program also predicts blade flapping behavior. The correlation with experimental results is generally good, i.e. within ± 0.5 degrees, which is understandable since the 2-D steady lift coefficients (Cl) used are less subject to errors than the related Cd values.

1.1.2 Prescribed Wake Models

The rotor wake calculation may become the central part of a rotor performance prediction program. An example is the UTRC program [6] which computes the induced velocities from the wake positions and the circulation around the blades. Figure 4 outlines the procedure for using the UTRC code to compute a rotor wake in a general rotor performance calculation which includes blade loads, stresses and vibrations.

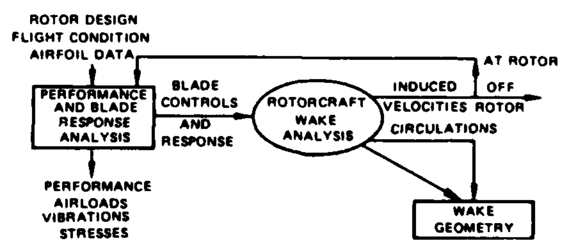

Fig. 4. Schematic of rotorcraft wake analysis (UTRC code), from [6]

This model has been used by SIKORSKY in a version in which the wake geometry is prescribed in the form of a skewed helicoidal sheet [7]. Each blade is represented by a curved lifting line which coincides with the bent quarter-chord line and the helical wake is modelled by discrete segmented vortex filaments. The comparison between computed and measured full-scale wind tunnel test results (figure 5) is interesting in that SIKORSKY attributes the discrepancies to the fact that dynamic stall effects were neglected, and to the highly simplified modelling of the tip sweep effect. In the example presented, the computed rotor power requirement is always underestimated (sometimes by as much as 5 %).

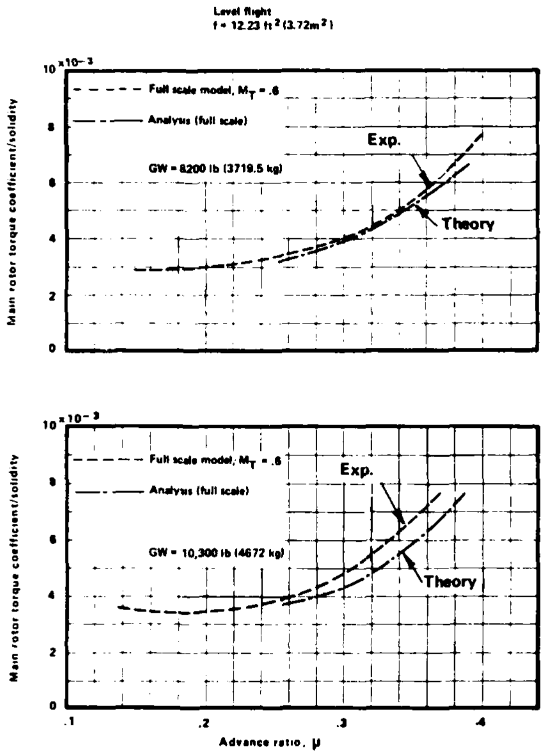

Fig. 5. Main rotor torque vs. advance ratio (SIKORSKY test vs. theory), from [7]

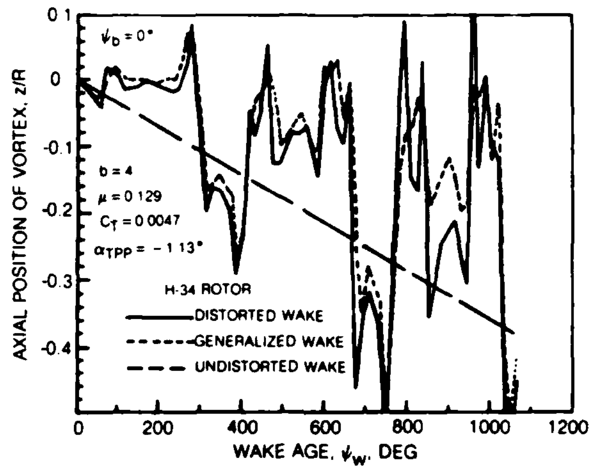

Fig. 6. Comparison of H-34 distorted and generalized wake geometries (UTRC code), from [8]

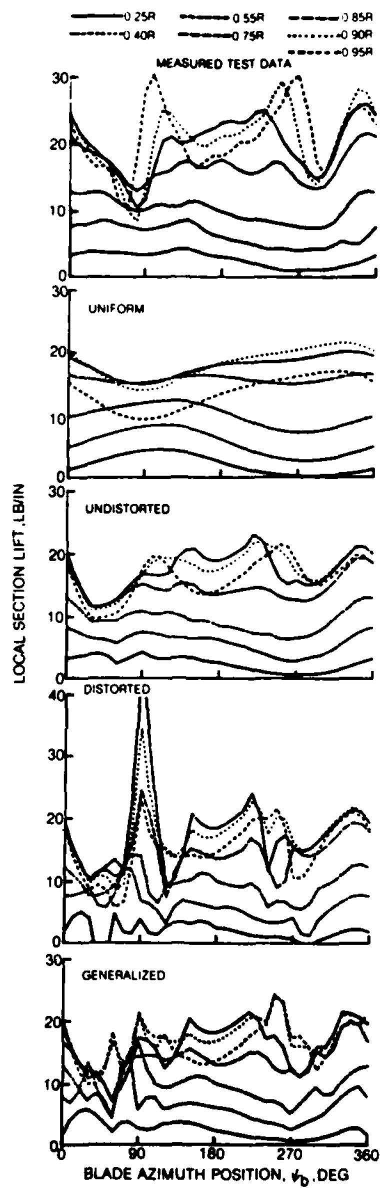

Fig. 7. H-34 lift distributions at μ=0.129, test/theory comparisons, from [8]

UTRC continues to work on wake models in an effort to synthesize their geometry, especially at low advance ratios. Many parameters are involved: rotor lift, attitude, blade number, aspect ratio and twist. Figure 6 shows the complexity of the distorted wake geometry of a four-bladed rotor at μ= 0.129. Figure 7 shows the effect of different wake models on the computed lift distribution and compares these with measurements made some 20 years ago [9]. Concerning the rotor power predictions obtained with the UTRC code, it is surprising to note that they are relatively independent of the wake model used despite the large differences between the computed lift distributions.

H34 rotor horsepower prediction (SHP)

Inflow model	μ=.064	μ=.129	μ=.290
Uniform inflow	635	468	511
Undistorted wake	681	499	543
Distorted wake	645	447	538
Generalized wake	690	468	542

The calculation of the velocities induced by a realistic rotor wake is generally very time-consuming. In order to reduce computing time the RAE has built a simplified wake model with vortex rings placed at prescribed distances below the rotor which seems to provide good correlation with measured blade loads [10].

1.1.3 Free Wake Models

This is the true physical problem, since no a priori assumption is made on the wake geometry or on the tip vortex positions which can be of considerable importance in certain rotor configurations particularily at low airspeeds. The free wake technique first developed by SCULLY [11] was used by W.JOHNSON in his Comprehensive Analytical Model of Rotorcraft Aerodynamics and Dynamics (CAMRAD) [12]. Figure 8 illustrates the application of this method to the determination of the lateral flapping of rotor blades at low advance ratio.

The CAMRAD code computes the wake by successive interations beginning with a "uniform" induced velocity estimate, then a prescribed wake, terminating with a free wake calculation as outlined in figure 9. Although the CAMRAD code attempts a comprehensive approach to the phenomena involved, it uses a lifting line model which has definite limitations near the blade tips where strong 3-D effects occur requiring a "lifting surface" approach.

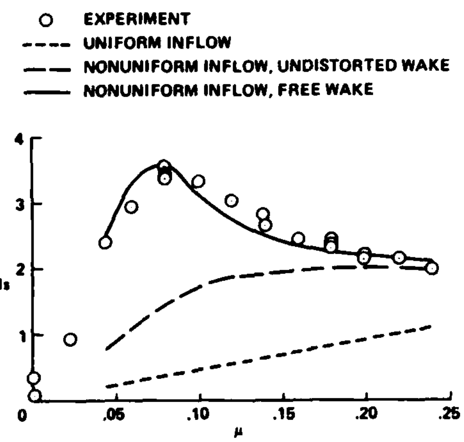

Fig. 8. Comparison of measured and calculated model rotor lateral flapping angles (CT/σ=0.08, αTPP=1°), from [12]

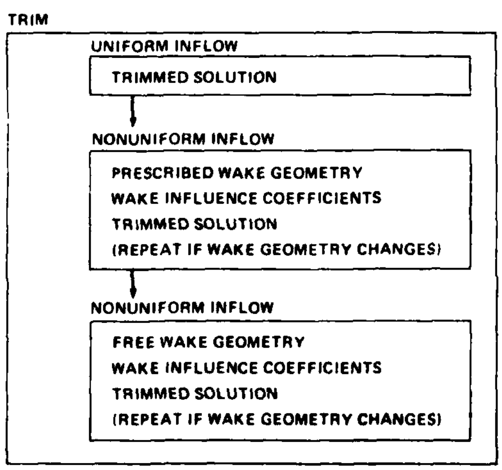

Fig. 9. Inflow analysis in CAMRAD for trim task

1.1.4 Lifting Surface Models

A computer code using the acceleration potential method was developed by ONERA and applied to rotary-wing aircraft [13]. It assumes unsteady 3-D linear aerodynamics but is presently only implemented for a blade lifting line and the wake is a prescribed helical sheet. Figure 10 shows a comparison with local lift measurements obtained on an AEROSPATIALE rotor (with rectangular blade tips) at the Modane S1 wind tunnel: very satisfactory agreement is obtained up to the blade tip regardless of the azimuthal position. ONERA is presently extending the method to blades of arbitrary planform by implementing a lifting surface formulation.

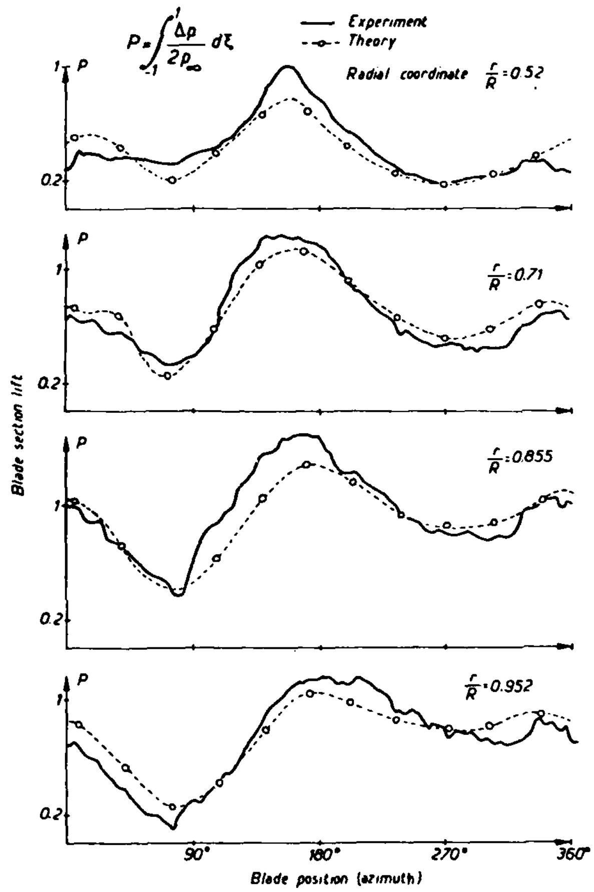

Fig. 10. Blade section lift vs. azimuth (acceleration potential code), from [13]

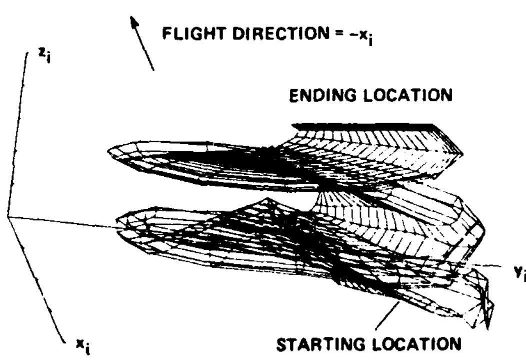

Fig. 11. Perspective view of the wake in forward flight (NASA singularities code), from [14]

NASA has developed a lifting surface unsteady vortex panel method for an isolated blade in forward flight which takes into account the mutual interactions between the blade and the wake [14]. Both the blade and the wake are broken down into panels. Figure 11 shows the inner wake distorsion and the formation of vortex flow at the blade tip. The first application of this lifting surface program in incompressible flow was made to study the anhedral effect of the blade tip.

ONERA is developing an "unsteady vortex point" lifting surface code [15] in which vortex particles are released by the blade and set in equilibrium as soon as they leave the blade trailing edges.

1.1.5 Dynamic Stall on the Retreating Blade

Dynamic stall of the retreating blade of a rotor is characterized by instantaneous airfoil lift values well above those achievable in steady conditions. Strong nose-down pitching moments generally produce large control loads, and, because of Cm-α hysteresis, energy may be transfered from the airstream to the blade resulting in possible stall flutter (figure 12).

Up to now, these configurations have mostly been studied in 2-D flow by oscillating the airfoil about the pitch axis in the vicinity of the steady-state stall angle of attack (heave motion has also been investigated). Considerable research has been done in this field, both in the United States (US Army RTL, BOEING-VERTOL, UTRC) and in Europe (ONERA, AEROSPATIALE, RAE, WHL). Many examples are presented in references [16] and [17].

The problem is to introduce the unsteady airfoil data in the rotor calculations. BOEING-VERTOL [18] and UTRC [19] have developed methods for synthesizing the experimental Cl and Cm values as a function of angle of attack α and of its time derivatives α° and $\alpha^{\circ\circ}$ whereas WESTLAND [20] chose to introduce a time delay in the stall calculation. Each of these methods improves the overall performance predictions, as shown on figure 13 illustrating the method used by BOEING-VERTOL [21].

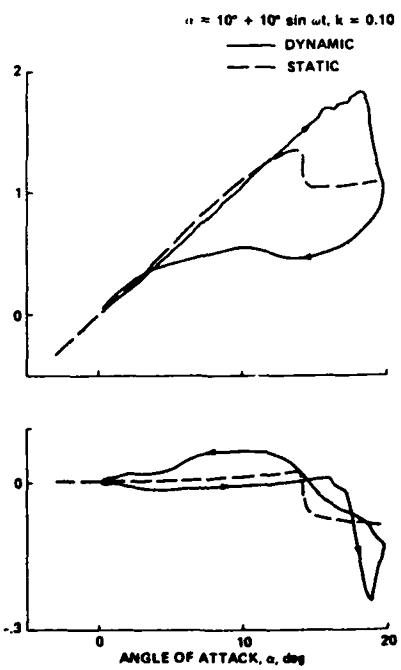

Fig. 12. Lift and pitching moment vs. angle of attack during airfoil pitch oscillations in a dynamic stall configuration, from [48]

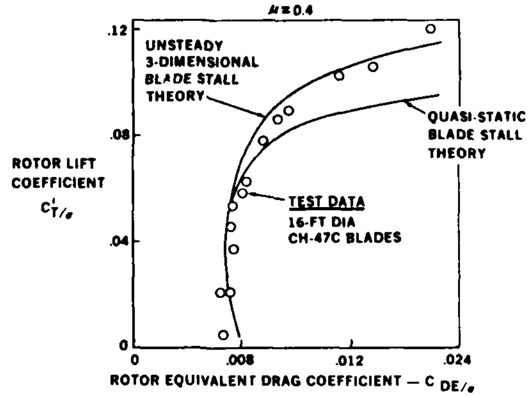

Fig. 13. Correlation between test data and BOEING unsteady blade stall theory, from [21]

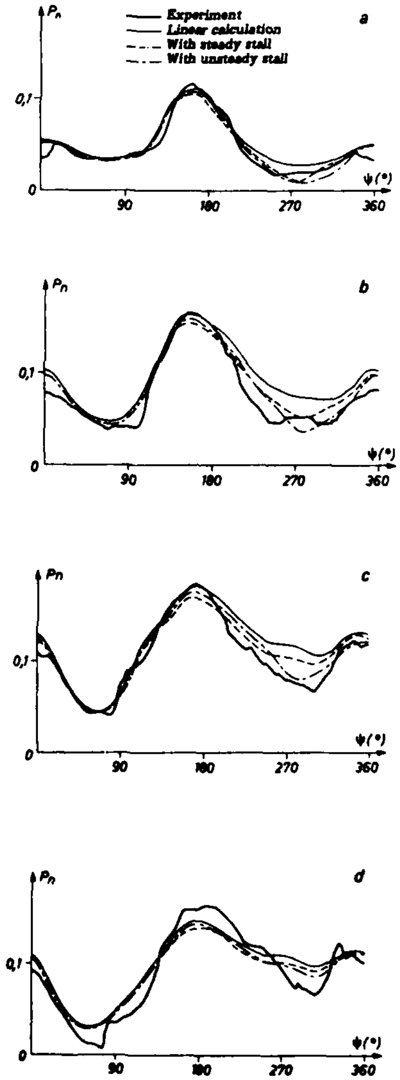

Fig. 14. Section lift vs. azimuth in stalled configuration (test vs. ONERA code prediction), (a) 0.52R, (b) 0.71R, (c) 0.855R, (d) 0.952R, from [22]

ONERA, has adapted its 3-D unsteady linear aerodynamics code (§ 1.1.4) to stalled airfoil configurations by defining an effective angle of attack which is the linear equivalent, in terms of lift, of the stalled airfoil angle of attack. By also incorporating the BOEING-VERTOL dynamic stall model, ONERA has now a much more realistic code for heavily loaded rotors on which retreating blade stall occurs. Figure 14 shows the good agreement between predicted and measured local lift values on the retreating blade.

These predictions should be further improved when the BOEING-VERTOL model is replaced by ONERA's recently developed phenomenological model of dynamic stall [23,24]. In this approach, the unsteady aerodynamic forces acting on an airfoil experiencing angle of attack variations are described by a non-linear second-order differential equation in which the coefficients are determined by the steady airfoil Cl(α) and Cm(α) curves and by the analysis of a limited number of cycles of low-amplitude sinusoidal pitch oscillations.

Ongoing work is now concentrated on modelling the effects of fluctuating attack velocities and 3-D flow based on tests of a wall-mounted half-wing with positive or negative sweep in unsteady flow. These tests are complementary to the experimental work of UTRC on the effect of airfoil sweep [25].

While lift and pitching moment effects are of primary importance as regards blade stresses and control loads, the drag factor must not be neglected because of its impact on the rotor power requirement [19].

1.1.6 Unsteady Transonic Flow on the Advancing Blade

This difficult problem has been examined for more than a decade by US Army RTL, ONERA, NASA, RAE and MBB. Since transonic flow primarily concerns the blade tips, it can only be correctly analyzed with a 3-D approach.

Until now, only the transonic small perturbation equation or the full potential equation have been used. While RAE [26] and NASA [27] initially studied a quasi-steady solution of the full potential equation (with the blade "frozen" in a given azimuthal position), US Army RTL and ONERA chose an unsteady solution of the transonic small perturbation equation. Figure 15 shows that in the simple case of a non-lifting rotor, the unsteady effect produced by the rising incident Mach number before the 90° azimuth position tends to delay the onset of shock waves, while, beyond the 90° azimuth, the decreasing Mach number is responsible for the appearance of strong shock waves on both rectangular and sweptback blade tips (which have an adverse effect on drag).

The first computer code developed jointly by US Army RTL and ONERA for an arbitrary blade geometry on a non-lifting rotor [28] was first adapted to lifting rotors [29], then extended to the calculation of the flow over the entire advancing blade sector (it was previously limited to the 90° azimuth position) [30]. In this calculation the rotor control positions and the blade flapping motion are prescribed and the simple DREES rotor inflow model is used. This latest version of the code was validated by comparison with ONERA test results on straight and sweptback blade tips defined by RAE and ONERA.

Figure 16 shows an example of very good agreement between computed and measured absolute pressure distributions on rectangular blades at three sections (0.85 R, 0.9 R and 0.95 R). The blades, with an aspect ratio of 7:1, were rigid enough to prevent significant deformation.

However a true prediction code should also compute the control positions corresponding to a given flight configuration. A decisive step in this direction can be achieved by matching a standard rotor code such as described in §1.1.3. with a program giving a detailed description of the flow over the blades. This has recently been attempted by NASA and US Army RTL [31], with an iterative procedure combining the CAMRAD code

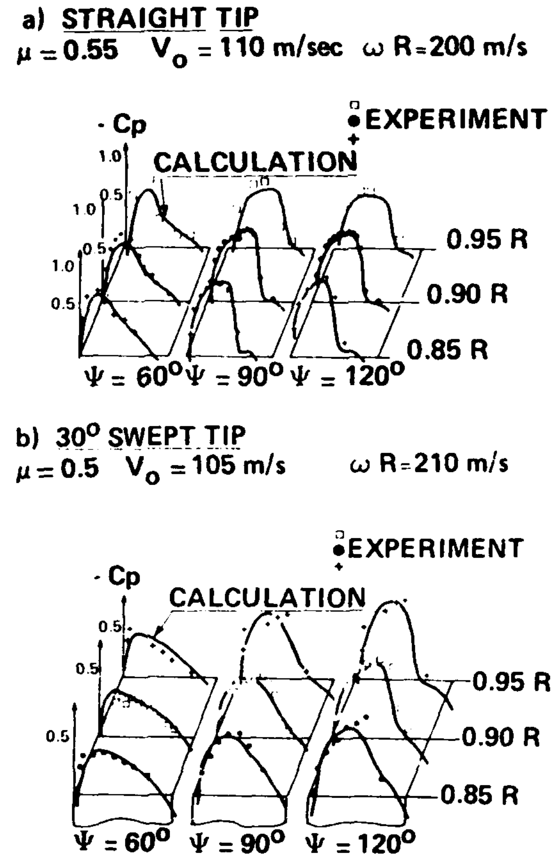

Fig. 15. Pressure distribution on non-lifting rotor blade tips, from [28]

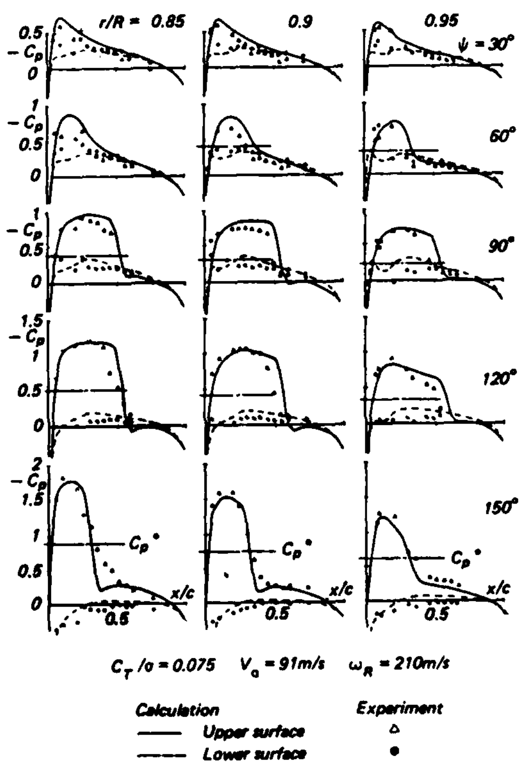

Fig. 16. Experimental and computed pressure distributions on rectangular blade tips, (ONERA TSP code), from [30]

(§ 1.1.3. and [32]) with a finite difference solution of the unsteady transonic small perturbation equation [29]. The matching method involves a transfer of lift, induced velocity and blade motion data between the two codes as illustrated in figure 17 from [31]. This amounts to replacing the standard 2-D airfoil data in the CAMRAD code by the local lift values determined from the unsteady 3-D flow calculation. Although the first configuration analyzed by this method concerned rectangular blades and was not very severe, some differences in the local lift values are already apparent and the calculated pressure distributions at r/R=0.95 are in relatively good agreement with the experimental data (figure 18). It would have been interesting to have some idea of the corresponding 2-D pressure distributions in order to appreciate wether or not the airfoil drag was liable to differ significantly from the 2-D value imposed in the CAMRAD calculation.

Rotor blade boundary layer integral methods have been developed by the DERAT in France [33]. The weak viscous-inviscid fluid coupling technique tested by ONERA gives a good physical representation of the viscous effects that can be expected in transonic flows (reference [30]). Unfortunately, it is not yet possible to make correct estimations of the aerodynamic drag of helicopter blades.

The need for specific drag calculations in 3-D unsteady flow is illustrated on figure 19 which shows pressure distributions at ψ=60° and 120° computed with the ONERA Transonic Small Perturbations Code [30]. Although the local lift values are very close for both azimuths, the corresponding chordwise pressure distributions are very different in spite of the fact that the local normal Mach number and angle of attack (1° in this case) are the same. It is sure that the drag and pitching moment of the three sections shown here will be very different at ψ=60° and ψ=120° and probably also quite different from the steady 2-D value at the same normal Mach number and Cl.

Even if 3-D rotor codes are presently incapable of correctly predicting the rotor torque they nevertheless offer a good description of the flow over the blade tips and give an indication of the potential hazards of using a particular blade shape (see [5,30,34] for example). Figure 20 shows how a sweptback parabolic tip designed by ONERA attenuates the maximum local Mach number in the advancing blade sector thereby reducing the rotor power requirement and the noise generated as verified by flight tests performed on the AS 365N.

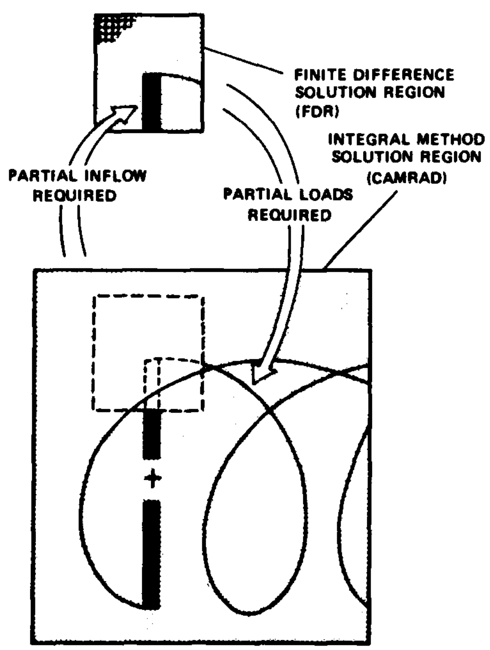

Fig. 17. Matching of integral and differential rotor flow methods, from [31]

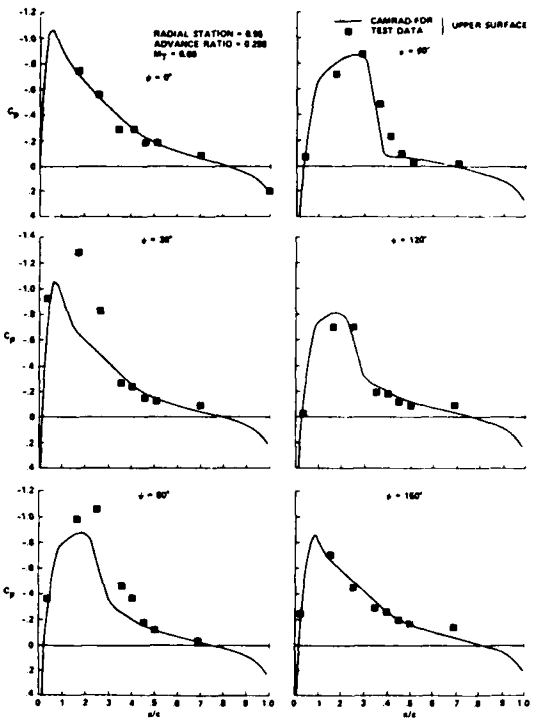

Fig. 18. CAMRAD-FDR computed pressure distribution vs. test results, from [31]

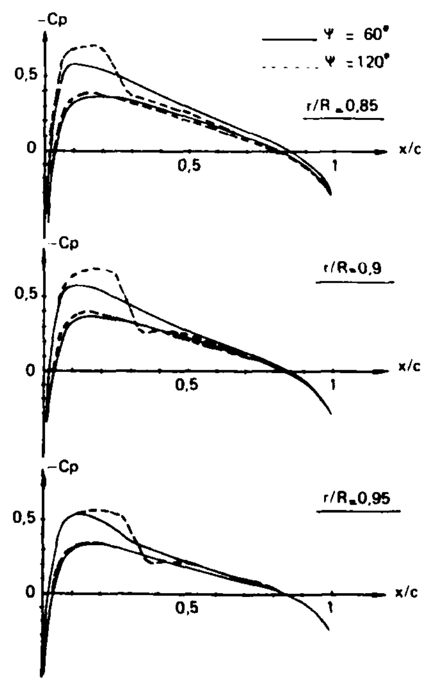

Fig. 19. Computed pressure distributions (ONERA TSP code, μ=0.4, $M_{\Omega R}$=0.64, α=1°)

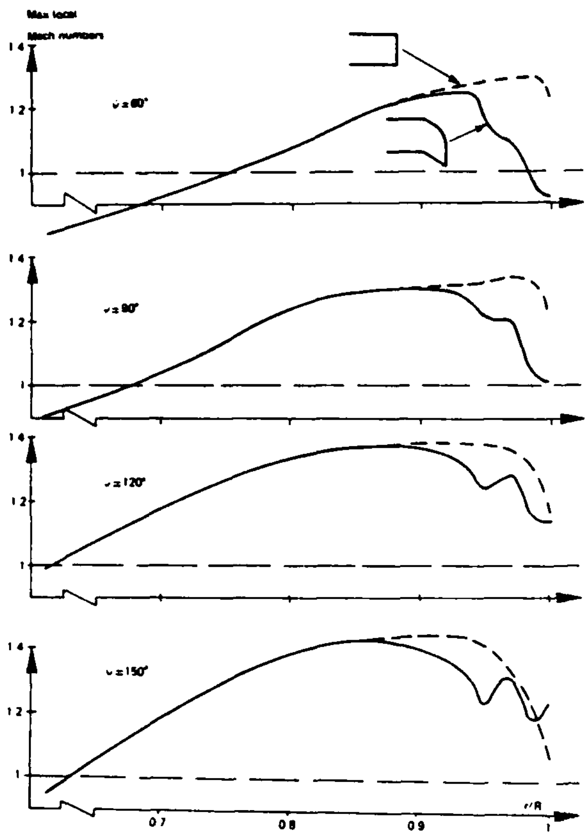

Fig. 20. Computed maximum local Mach numbers for rectangular blade tip and parabolic sweptback tip (ONERA code), from [35]

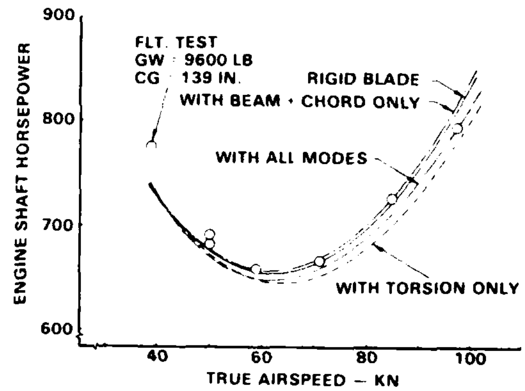

Fig. 21. BELL prediction of power required on BHT model 212, from [37]

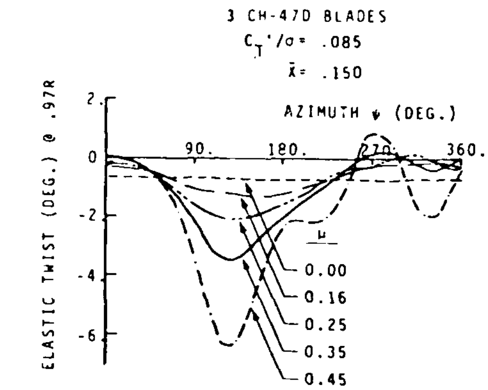

Fig. 22. Effect of advance ratio on model blade elastic twist at 0.97R, from [51]

1.1.7 Blade Deformations

The more flexible the blades, the greater the deformations due to aerodynamic and inertia forces which have to be taken into account in the calculation of the airfoil operating conditions. BELL has used the C81 program (Rotorcraft Flight Simulation Program [36]) to study the influence on the computed rotor power of various torsion, flapping and lead-lag bending modes. Figure 21 shows clearly that elastic twist has the largest influence on the final result probably because this mode produces a direct change in the blade element angle of attack. The figure also indicates that the power reduction due to the torsional deformations is offset by the flapwise and chordwise motions. It is therefore necessary to include at least the first three coupled torsion-bending modes in the calculation. We may also be tempted to conclude that the rigid blade assumption gives satisfactory results but it is most likely that this would turn out to be wrong at higher speeds or with "softer" blades than those of the BHT model 212 used in the case presented here. The modelling of at least the first torsion and flapping modes is certainly a necessity when dealing with modern rotor blades especially when these are fitted with swept tips. Another example of the importance of the elastic blade deformations can be found on figure 22 which shows the effect of advance ratio on the blade torsional motion. It is clear that such elastic twist effects (particularly in the advancing blade sector) have to be modelled in order to make a valid prediction of the rotor loads.

1.1.8 Fuselage Interaction Phenomena

Up to now we have only delt with the isolated rotor. The proximity of the fuselage introduces mutual interference effects which cannot always be neglected. The rotor wake interacts not only with the fuselage, but also with the aircraft tail section (stabilizer, tail fin, tail rotor). These problems will be considered in chapter 2; the following discussion will cover only the interference created by the fuselage on the main rotor.

Figure 23 shows that the fuselage flow field creates upwash in the front portion of the rotor disk and downwash at the rear and that these perturbations get stronger as the rotor is moved closer to the fuselage. They were computed by MBB for an isolated fuselage with a panel method. The effect of the fuselage is to increase the blade angle of attack significantly at ψ=180°, and to reduce it at ψ=0° (figure 24). The largest variation occurs at mid-span, in the forward blade position, where the local lift may be as much as 50% above the isolated rotor level.

Similar studies conducted by RAE [39] have shown that the fuselage interaction, which modifies the azimuthal variation of blade lift (and consequently the blade bending and torsion moments) and the rotor torque, also changes the vibratory forces and moments acting on the rotor hub and thus the aircraft vibration level.

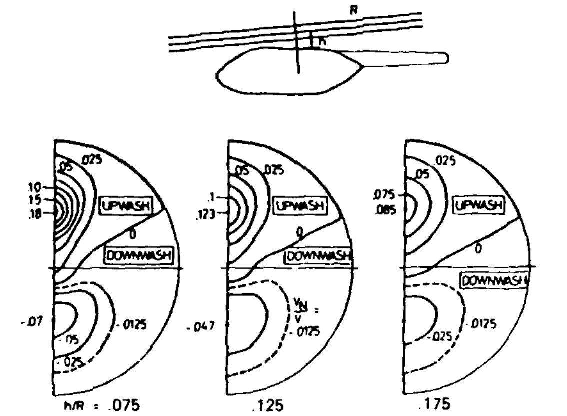

Fig. 23. Fuselage induced flow field in rotor plane (MBB calculation), from [38]

NASA has undertaken a systematic experimental and theoretical study of the aerodynamic interaction between the rotor and fuselage [40] which revealed that the rotor equivalent lift/drag ratio, which is generally lower for the rotor-fuselage combination than for the isolated rotor, could be improved if the rotor was correctly positioned relative to the fuselage. There is matter here for further investigation on the optimization of the rotor position so that performance improvements obtained on the isolated rotor remain perceptible in presence of the fuselage.

Finally, we should mention the efforts made by AMI [41] to achieve full coupling between the rotor and a realistic helicopter fuselage as shown in figure 86 (§2.2.2.3). Although the rotor in this case is treated as a lifting disk with a time-averaged wake, an attempt is made to compute the rotor wake distortions in the presence of the fuselage.

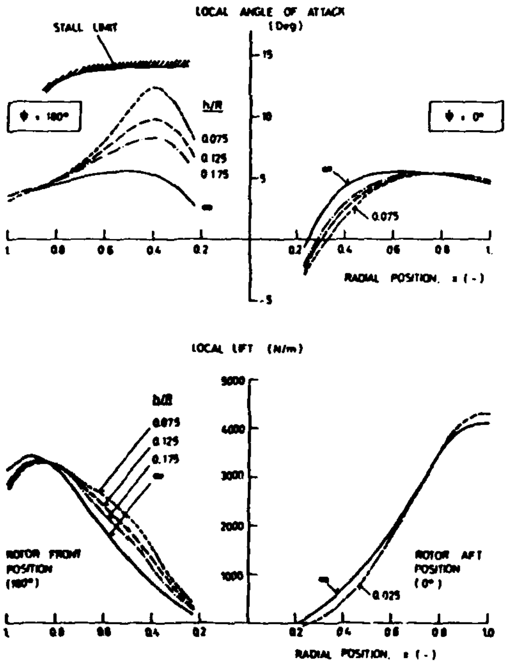

Fig. 24. Effect of fuselage flow on local angle of attack and lift at 150 knots, from [38]

1.1.9 The Rotor in Hover Flight

This configuration is critical, since it often determines the rotor and overall aircraft design. It is characterized by a very complex wake geometry as indicated on figure 25.

Modelling the wake, and especially the tip vortex, has long been and still remains a major problem. Prescribed wake techniques are still in wide use: the wake geometry is given by more or less empirical rules based on a large body of experimental data such as gathered by UTRC [42] and BELL [43] for example. The tip vortex trajectories depend on the rotor thrust, but also on the number of blades, their twist and aspect ratio. The prescribed wake technique was further improved by attempts to correlate the wake geometry with the circulation distribution as in SIKORSKY's Circulation Coupled Hover Analysis Program (CCHAP) [6]. It has been shown for example that the tip vortex trajectory was directly related to the maximum blade circulation value. This has made it possible to extend the validity of the prescribed wake approach to rotors with non rectangular blades (e.g. trapezoidal blades as shown in figure 26.

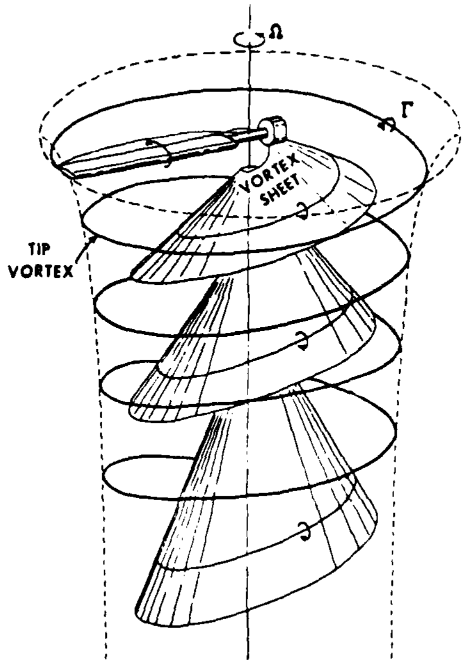

Fig. 25. Schematic of hovering rotor wake structure

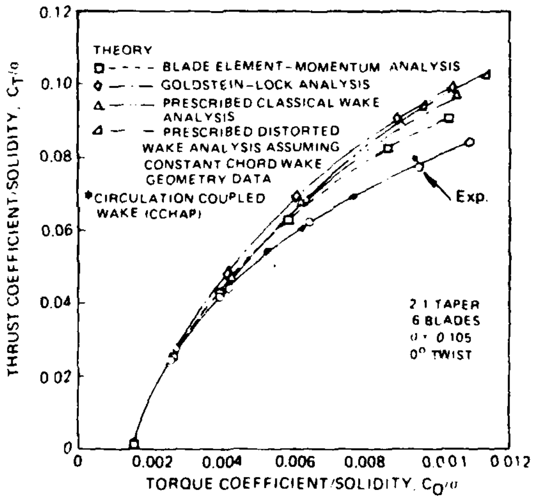

Fig. 26. Correlation of CCHAP with test data on a model rotor with tapered blades, from [6]

However, a more rigorous approach, despite the longer computer time required, can only be obtained by calculating the complete or partial wake equilibrium under the influence of the rotor-induced velocities. With regard to the blade itself, either a lifting line theory, as used by AEROSPATIALE [44] or NASA [12], or a lifting surface theory, as chosen by BELL [43] or AMI [45,46], may be adopted.

Moreover the prescribed wake approximation is often the first step in a free wake calculation. For a non-rectangular blade such as the one proposed by BINGHAM in [47] (trapezoidal with a 1:3 taper ratio initiated at 0.5R), calculations made by SUMMA of AMI [46] show that the tip vortex trajectory is very different from the one given by the LANDGREBE and KOCUREK formulas (figure 27). The distribution of circulation along the blade is very sensitive to the wake and tip vortex positions as can be seen on figure 28, and, for a given collective pitch setting, convergence is not obtained for the same value of rotor lift (the discrepancy exceeds 10% in this case).

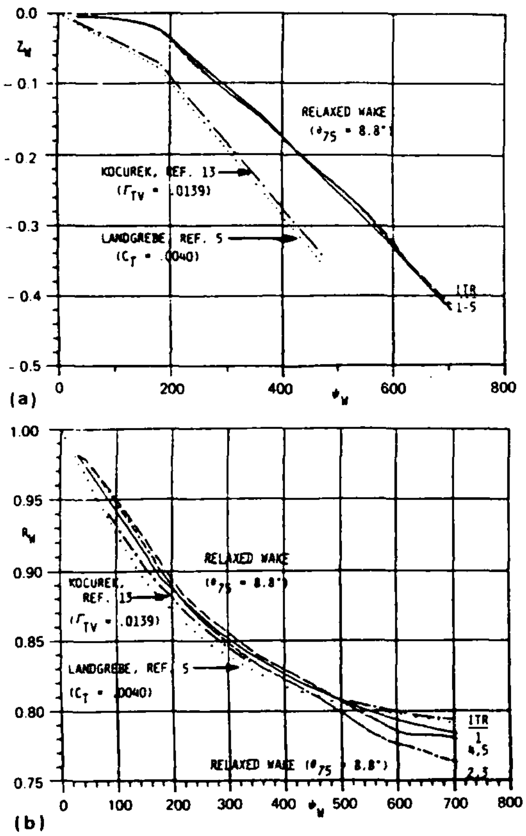

Fig. 27. Comparison of prescribed and relaxed tip vortex geometries on the BINGHAM rotor (AMI code, (a) axial, (b) radial coordinates), from [46]

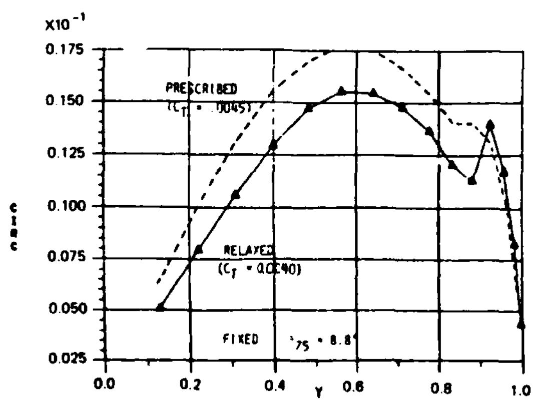

Fig. 28. Prescribed and final relaxed wake circulation on the BINGHAM rotor (AMI code), from [46]

As reported in [48] Dr. TUNG of US Army RTL has made an interesting review of several computational codes (figure 29) which were applied to a rotor with low aspect ratio blades (λ=6) for which measurements were available both on the blades and in the wake [49]. It can be seen that satisfactory agreement between computed and experimental results was obtained only by combining a lifting surface approach with a free wake model for which the equilibrium position was also in agreement with the experimental data. However, Dr. TUNG indicates that a similar agreement could not be achieved in the case of a rotor with highly non-linear blade twist.

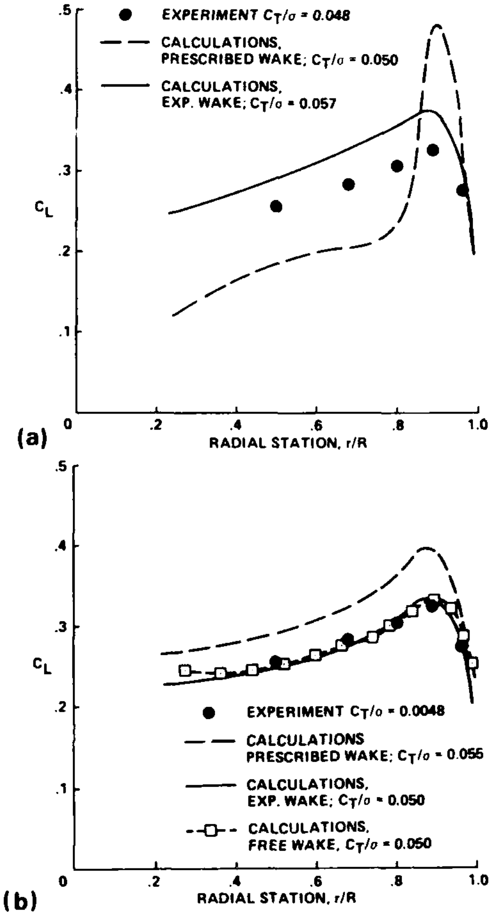

Fig. 29. Spanwise lift distributions in hover (US Army computation, λ=6, θ=8°, (a) lifting-line code, (b) lifting-surface code), from [48]

One of the remaining unsolved problems concerns the actual formation of the tip vortex on the blade and the roll-up of the vortex sheet released at the blade trailing edge. Figure 30 shows the techniques used by AMI [46] in two of their programs. ONERA is presently developing and validating an "unsteady vortex point" method [15] based on the resolution of HELMHOLTZ's vortex equation which ensures that the vortices naturally reach their equilibrium positions after they leave the edge of the blade.

The wake geometry determines the velocity field through the rotor disk and consequently the local angle of attack, the aerodynamic lift and the associated drag. It should be borne in mind that approximately 2/3 of the rotor power required in hover is due to the rotor downwash while the remaining 1/3 is necessary to overcome the profile drag. The local lift and drag forces are generally obtained from 2-D airfoil data. Even if this problem appears less critical than in forward flight, progress could be achieved by associating the rotor downwash calculation (using one of the singularity methods mentioned earlier) with a more detailed blade pressure distribution calculation. This has already been attempted by US Army RTL and ONERA [29] by

combining an AMI "Hover" code with a "Transonic Small Perturbations" code, giving encouraging results with regard to experimental data. This technique seems to be necessary when dealing with highly loaded rotor configurations in which transonic flows are liable to be present over the blade tip airfoil sections. In this respect ONERA has recently developed a full potential code to replace the transonic small perturbations code which has definite limitations at high lift.

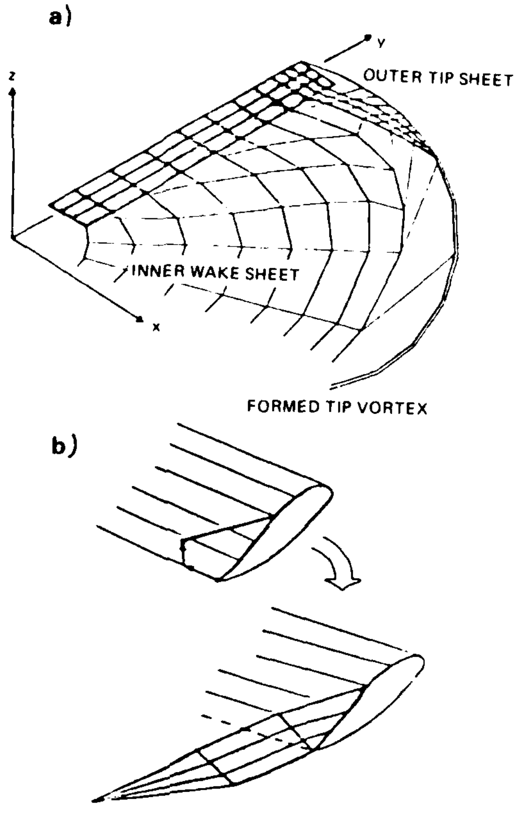

Fig. 30. AMI models of tip wake geometry (a) Hover code, (b) Rotair code, from [46]

Figure 31 illustrates the problem of premature airfoil stall at low Reynolds numbers. At 1:5 scale the performance of the rotor with tapered blades drops off at high lift when compared to the rotor with rectangular blades, whereas no such difference is noted in the full scale tests. In addition, the performance characteristics measured on the 1:5 scale model are appreciably lower than those measured at full scale (the figure of merit is decreased by about 0.08). The reason for this is the increase of airfoil drag at low Reynolds numbers.

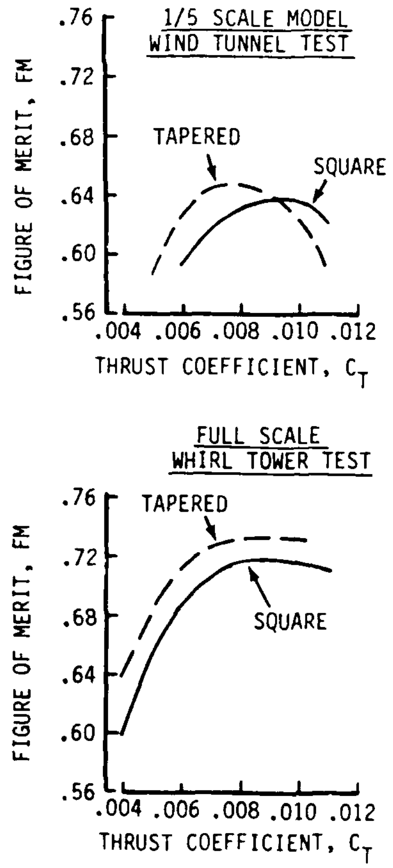

Fig. 31. YUH-61A model and full-scale rotor performance in hover with rectangular and tapered blade tips, from [51]

1.2 EXPERIMENTAL METHODS

It has been shown that numerical methods are still far from being capable of accurate rotor performance predictions and that testing remains necessary to define the limits of a new theoretical approach or to check the validity of a new concept.

1.2.1 Wind Tunnel Tests

Wind tunnel tests have the advantage of allowing accurate measurements under controlled conditions, but are generally limited to reduced scale models, with the notable exception of NASA's Ames facility which is capable of full scale rotor tests.

The subject of wind tunnel testing has been thoroughly covered by F. HARRIS in the AGARD Lecture Series N° 63 [50], and only a few topics where progress has been made will be considered here.

1.2.1.1 Rotor Performance Estimates based on Scale Model Tests

At reduced scale, the Reynolds number is generally lower than for the full scale rotor even when Mach number and advance ratio similarity is observed.

BOEING-VERTOL [51] has developed a technique for estimating the performance of a full size rotor from scale model tests based on studies of the effect of Reynolds number on steady and unsteady airfoil data. This method allows BOEING-VERTOL to show satisfactory correlation between corrected wind tunnel data and flight test results for the CH-47D rotor both in hover and forward flight, as shown on figure 32.

The difficulty of making such comparisons should not be underestimated. It is not easy to determine accurately the rotor lift in forward flight because of fuselage and stabilizer download; similarly, the rotor propulsion force is difficult to evaluate since the exact aircraft drag can only be estimated. A recent study made by SIKORSKY [7] is interesting in this respect since it shows a comparison of S76 flight test data with 1:5 scale model test results of the complete helicopter and with full scale isolated rotor tests made in the AMES wind tunnel. Figure 33 compares the rotor performance of the three configurations and shows once again that the uncorrected small scale rotor leeds to pessimistic estimates of the full scale rotor performance.

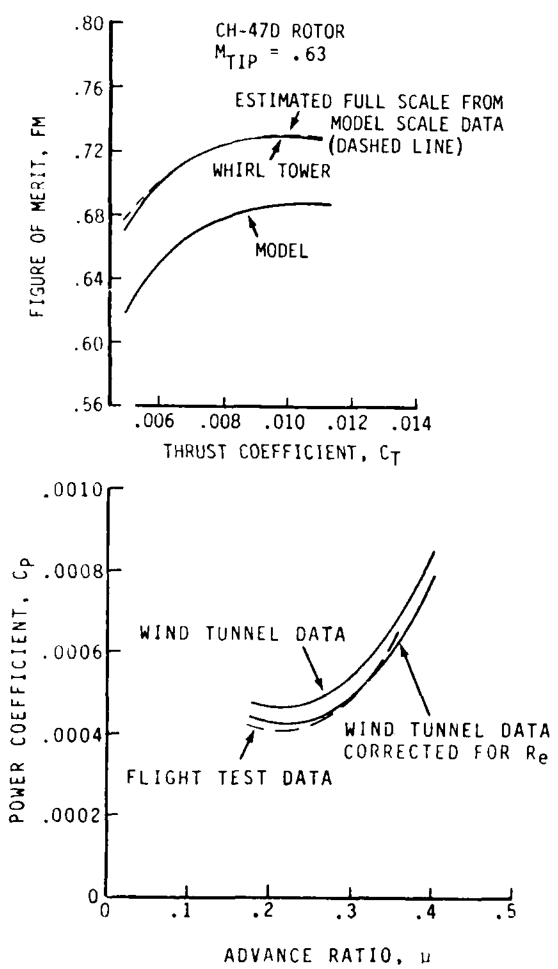

Fig. 32. CH-47D full scale performance estimation based on model test data (BOEING calculation), from [51]

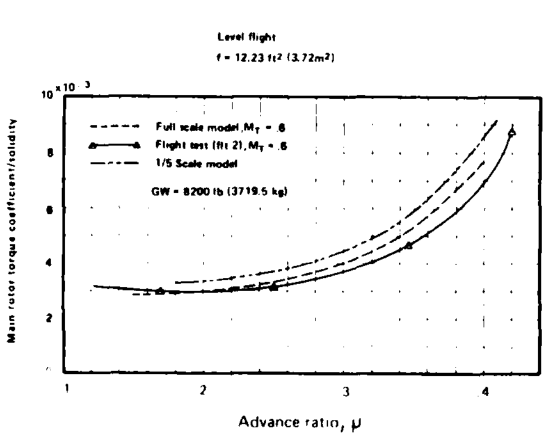

Fig. 33. Main rotor torque vs. advance ratio (flight/wind tunnel test comparison on S-76 rotor), from [7]

Nevertheless, model rotor tests are usefull when comparing different rotor geometries, at least for standard configurations, and when validating theoretical methods which take into account Reynolds number effects. Scale models are also much less expensive to build, and can be tested in configurations that would not ordinarily be possible in flight (e.g. at very high speeds) or that would involve unacceptable risks. For example, BOEING-VERTOL recently tested a series of 7 different blades (figure 34) on three and four bladed rotors measuring 10.1 feet in diameter at airspeeds up to 230 knots. A systematic investigation of the effect of new airfoils, blade planform and aspect ratio on rotor performance was made and the results published in [52]. Figure 35 shows the influence of blade shape on rotor efficiency in forward flight. A trapezoidal tip with a 1:3 taper ratio initiated beyond 0.9R was found optimal (and in hover as well). The extrapolation of these results to a full scale rotor using the technique presented in [51] comfirms the advantages of this geometry as shown in figure 36. The BOEING-VERTOL B-65 code described in [54] appears to predict these experimental results correctly both for the rectangular and the trapezoidal blade tips.

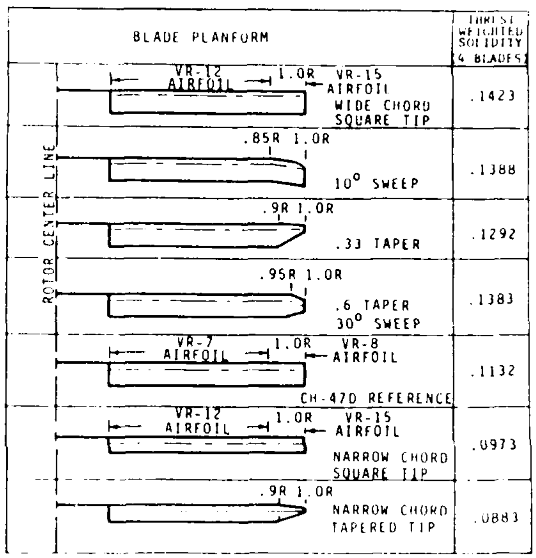

Fig. 34. Blade configurations tested by BOEING-VERTOL, from [52]

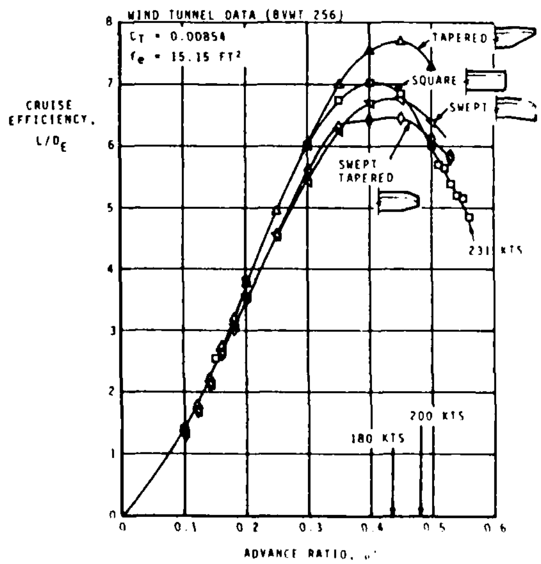

Fig. 35. Effect of tip shape on cruise efficiency of BOEING rotor models, from [52]

The fuselage afterbody is thus surrounded by a complex flow system which can undergo complete transformations accompanied by large variations of drag in response to small changes of angle of attack. Unfortunately, as will be discussed later, analytical methods are presently unable to deal with separated flow and experimental methods will no doubt continue for a long time to be the basic tool used in the evaluation of fuselage drag especially when operational requirements impose unfavourable geometric constraints on the afterbody (strong contraction or camber). In order to obtain valid drag predictions for preliminary design at the project stage, wind tunnel parametric data sheets such as those reported in [106,115] are extremely valuable.

Rotor Head Drag

Like the afterbody section, the rotor head is a problem area for the aerodynamicists. It generally accounts for 20 to 30 % of the total drag [138] but, in the case of a streamlined fuselage with no similar improvement of the rotor head, this proportion may rise to 50% [118]. Clearly, a substantial amount of work remains to be done here as well.

It was mentioned earlier that rotor head drag measurements were difficult, and apparently sensitive to the effect of Reynolds number. The problem of the representation at small scale of rotor head models was also noted. Comparisons of wind tunnel data of rotor heads tested under different conditions must therefore be made with caution, especially since it is difficult to isolate completely the rotor head from the fuselage: the rotor head drag may double depending on whether it is measured alone or whether it is measured in the presence of the fuselage (or at least of the rotor mast fairing) [138,119]. It is not an easy matter to decide where the rotor head actually ends. On the HPER [99] the forces and moments are measured on the rotor head, as well as on the rotor mast fairing, as the final objective is to reduce the overall drag and not simply the drag of one component at the expense of the other. Similarly, drag must be measured without the rotor blades. But where is the dividing line between the hub and the blades? The blade neck is generally included in wind tunnel tests on small models, thereby significantly increasing the drag of the rotor head alone [104] On full scale models however the blades are removed [95,100] and the rotor head extends only as far as the hub shanks.

Another problem is that all rotor heads are not alike. A distinction has long been made between articulated and rigid hubs, the geometry of the former being more complex and the drag higher than for the latter. Faired hubs could also be distinguished from unfaired hubs. Today, however, new concepts are being considered with entirely new geometries [120] for which available wind tunnel test results can only be extrapolated with extreme caution. For example [96] illustrates the general trend of rotor head drag to rise with aircraft weight. In [120], the highest and lowest drag values of the hubs tested differed by as much as a factor of 5. The statistical trend is thus insufficient and carefull attention must be given to the rotor head design. It is thus difficult to provide general results for rotor head drag, and virtually every new hub design requires a new drag study.

Nevertheless, it is an accepted fact that for a given type of hub the drag increases with the hub size [96,109,119]. However, carefull consideration should be given to the definition of the reference area used to measure the hub size; there is no precise and universal definition of the frontal area of a rotor head. Some authors [95,96] consider the cross sectional area of the volume of revolution generated by the rotating head which exagerates the importance of the pitch-control rods and increases the reference area. Others, [109], use the maximum projected area of the stationary rotor head, which gives a much smaller reference area and therefore higher drag coefficients. It would certainly be useful if the rotor head drag data could be referenced to both areas, since the permeability of the rotor head could then be taken into account (the presence of "holes" is generally detrimental from the standpoint of drag).

One solution for reducing rotor head drag is therefore to design a more compact hub. The size reduction is directly favorable and, in addition, a smaller hub will certainly be less permeable. But this is not primarily an aerodynamic problem so much as a technological one.

Another possible solution is to design a fairing around an existing hub to reduce its drag coefficient rather than its physical size. Numerous hub fairings have been tested in wind tunnels, and occasionally in flight. Some were efficient [107,121], while others proved disappointing [122] A rotor hub fairing implies a larger cross section, which, in some cases, can offset the gain due to streamlining. To avoid this problem the rotor head should be designed to match the fairing rather than attempting to improve an existing unfaired design. Another difficulty has been raised in [109,137]: the fairings tested were found to increase drag until they were made airtight thereby suppressing internal airflows which are detrimental to drag. Today this is acknowledged as a prerequisite to the effectiveness of a fairing, but airtightness is difficult to achieve because of the large cut-outs and seals necessary for the passage of the blade shanks throughout their travel ranges. Figure 74 from [120] shows an attractive solution in which the fairing is an integral stressed component of the hub rather than a simple adjunct. The claimed hub drag (0.15 m^2 for a helicopter weighing over 7 tons) is far below the usual values.

Despite some interesting possibilities, hub fairings are nevertheless seldom installed on production helicopters; the most important obstacles concern aircraft maintenance and servicing rather than aerodynamics.

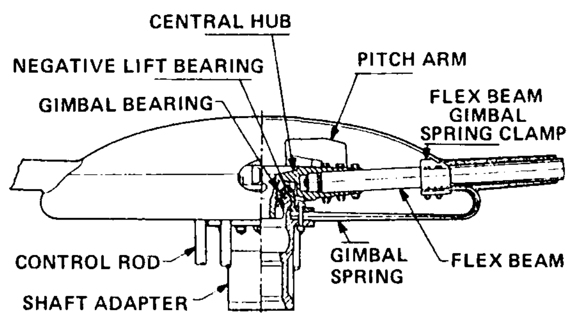

Fig. 74. Low drag hub with stressed fairing, from [120]

2.1.3.2 Fuselage and Rotor Head Wake Interaction Effects

Interactional aerodynamics covers an important area of helicopter aerodynamics [108] dealing with the complex interactions between main rotor, tail rotor and fuselage airflows sometimes involving ground effect. The following discussion will be limited to a description of the fuselage and rotor head wake effects on the tail fin and horizontal stabilizer which can lead to two types of problems:

- Fuselage static instability due to a loss of aerodynamic efficiency of the tail surfaces operating in the wake generated by the fuselage and rotor head
- Tail shake, a dynamic flow instability characterized by the onset of intermittent low-frequency vibrations of the tail structure (tail boom, fin or stabilizer).

Fuselage static instability and tail shake are the static and dynamic effects of the interactions considered here.

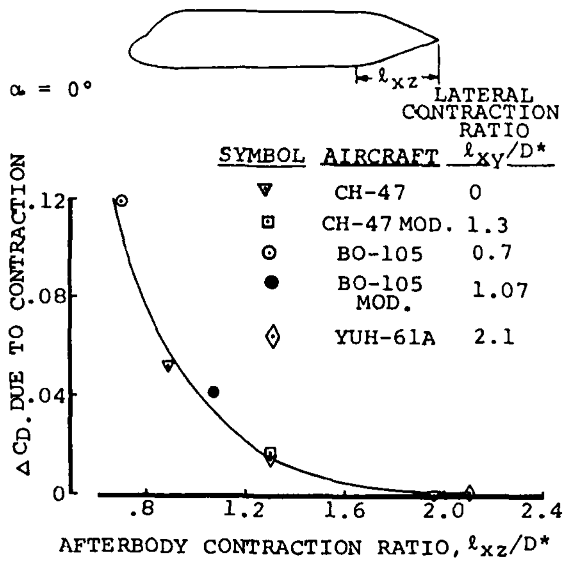

Fig. 71. Fuselage drag trend with afterbody contraction, from [111]

Figure 72 shows the effect of another parameter which is characteristic of the shape of the aft fuselage: the camber. At zero angle of attack, a highly cambered afterbody does not incur a large drag penalty over a symmetrical (uncambered) section but the two shapes behave very differently with regard to angle of attack.
At angles corresponding to cruising flight ($\alpha \simeq -5°$) the drag of the fuselage with the symmetrical afterbody changes very little, while a considerable increase is observed on the fuselage with a cambered afterbody. Wind tunnel tests made by AEROSPATIALE confirmed this trend and indicated that a fuselage with a highly cambered afterbody and a strong cross-sectional area contraction, could have, at -5° angle of attack, approximately twice the drag it had at zero incidence.

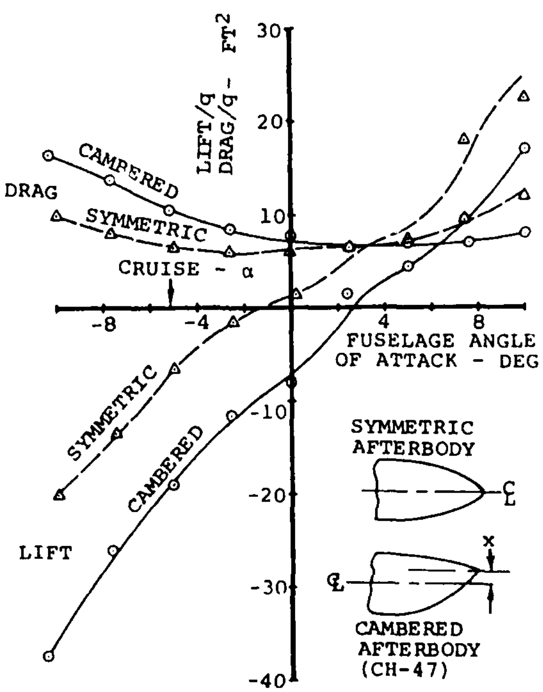

Fig. 72. Effect of afterbody camber on fuselage lift and drag, from [111]

The fuselage afterbody has thus a critical effect on drag. However operational requirements (such as the requirement for a rear door) may impose a significant contraction or camber of the rear section despite the drag penalty. Recent attempts to measure the influence of the fuselage afterbody shape on drag and to describe the physical phenomena involved can be found in [106,152,115,116].
SEDDON in [115] has revealed the existence of two types of flow conditions depending on the value of the angle of attack and the inclination of the afterbody axis. Figure 73 plots the fuselage drag versus the angle of attack. As the negative (nose-down) angle of attack increases to a critical value ($\alpha \simeq -9°$ on figure 73) the drag suddenly increases and the nature of the flow changes, as shown by visualisation tests with wool tufts and pressure measurements on the fuselage afterbody. Below this critical angle, the flow pattern is the classical flow which can be observed behind bluff bodies with a steady wake (eddy flow). When the critical angle of attack is reached, two vortices appear on either side of the fuselage resembling those generated by a low aspect ratio lifting wing (vortex flow).

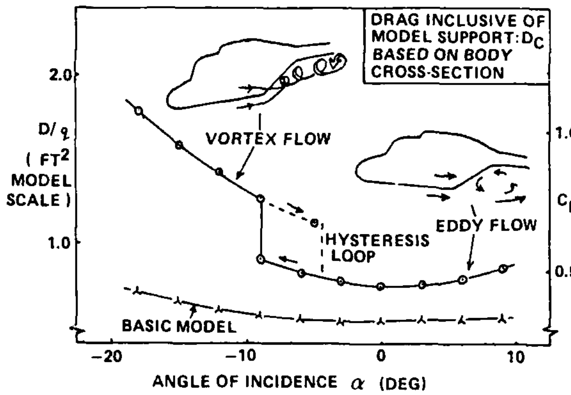

Fig. 73. Eddy flow/vortex flow transition: effect on fuselage drag, from [115]

The critical angle of attack value depends on the inclination ϕ of the afterbody axis which allows SEDDON to plot a boundary in the (α, ϕ) plane separating the region of eddy flow from the region of vortex flow.
A similar flow transition has also been noted at the rear of automobiles in [117] for example. In this case the incidence remains zero, and the inclination ϕ of the rear face is varied. When ϕ rises up to about 30° vortex flow conditions prevail and the drag progressively increases. The transition to eddy flow occurs betwen 30° and 35°, and the drag stabilizes at an intermediate level between the drag at very low ϕ and the maximum drag.

In [116], SEDDON suggests a number of solutions to reduce fuselage drag by forcing the flow to remain an eddy flow down to larger negative angles of attack. The addition of a spoiler delays the transition from eddy to vortex flow from -4° to -10° angle of attack, thus allowing considerable drag reductions at cruise attitudes. This result corresponds to the measured influence of the spoiler installed on the BO 105 in [111]. But perhaps the most promising solution is to add a row of small deflectors on each side of the afterbody, thereby completely preventing any transition to vortex flow throughout the angle of attack range (- 15° to + 10°). This solution makes it possible to design an afterbody section capable of accommodating a rear loading door without incurring the severe drag penalty due to the flow transition.

Vortical flow was also observed in [152] which reports on flow measurements in the wake of a fuselage with an afterbody having a high inclination angle (ϕ=41°). Two large counter-rotating vortices with longitudinal axes were identified, similar to those described in [115].

Another related problem concerns the airtightness of the model. Any slit that allows air to pass through will increase the drag, both on the aircraft and on the model. The drag penalty depends on the leakage flowrate of the model which must therefore be determined. If it cannot be measured it is preferable to use an airtight model and to correct the results according to an estimation of the leakage flowrates on the full scale aircraft rather than to measure drag with an unknown flowrate. Care is thus required to prevent any spurious leakage of the model by making it airtight.
The drag due to gaps between the cowlings and to protruding elements (antennas, lights, etc.) was measured at full scale in [95] and was found to represent between 9 % and 13 % of the total fuselage drag depending on the configuration tested.

2.1.2.5 Problems Specific to the Rotor Head

The rotor head plays a major role in the aerodynamics of the helicopter. It makes a large contribution to the total aircraft parasite drag and appears to be more sensitive to the effects of Mach and Reynolds number than the fuselage, as noted earlier. It was also stated that the rotor head geometry was difficult to represent exactly at model scale, and that these errors could lead to inaccurate drag estimations (§2.1.2.1). Fortunately, the other parameters seem to have little effect on the results, and it is correct to assume that rotor head drag is not significantly altered by the fuselage angle of attack and sideslip or by the rotation speed of the hub (see for example [96,99,104,105,106] for more details).
The fact that rotor head drag is relatively independent of the fuselage angle of attack and sideslip (as long as these are kept small i.e. within $\pm 8^\circ$) is due to the airflow in the vicinity of the rotor head which is deflected by the fuselage so that the local angle of attack changes very little with the upstream flow conditions. Reference [105] shows that this is true only as long as the main rotor shaft tilt angle with respect to the fuselage remains small (within $\pm 5^\circ$), which is the case for existing helicopters.

The independence of drag with respect to the rotational speed of the head is commonly accepted and allows wind tunnel tests to be conducted at any rotation speed or even with the rotor head at rest. SHEEHY in [96] restricts this statement to unfaired rotor heads, because test results presented in [93] seemed to indicate an increase of the drag of faired rotor heads with increased rotational speed (however this result was not consistent throughout the Mach number range covered and no definite conclusions can be drawn). Conversely, reference [105] showed that the aerodynamic coefficients of several faired rotor heads were practically independent of their speed of rotation when the blade shanks were not represented. When the shanks were included, the aerodynamic coefficients plotted versus the advance ratio μH (relative to the rotor head radius) varied at low μH but reached an asymptotic value before $\mu H=5$. However the blade shanks tested were in fact airfoil sections, i.e. highly streamlined, and not representative of existing designs.

2.1.3 Test results

The first helicopters were flown without prior wind tunnel testing of the fuselage, but today several wind tunnel test campaigns are conducted on different models before the aircraft is built. For example, four models were used in designing the EH 101 [106]:

- a 1:7 scale model for the design of the cowlings, rotor head, radome and sponsons, to evaluate aircraft stability and for streamlining
- a 1:4.5 scale model for air intake testing
- a 1:12.5 scale model complete with rotor for studying air recirculation flows, longitudinal and lateral stability, drag breakdown, fuselage surface pressure measurements and rotor/fuselage interactions
- a 1:3.2 scale model of the rotor head for rotor head drag reduction and cowling surface pressure measurements.

Most of the problems liable to arise in flight are thus first analyzed in the wind tunnel to avoid major, and extremely expensive, design changes during the development phase. As wind tunnel tests become more frequent they are also becoming increasingly complex. After global measurements of aerodynamic forces and moments acting on the fuselage with a 6-component balance, the models were progressively equipped with multiple balances (3 in [107], 5 in [108]) to weigh different aircraft components separately (main rotor, fuselage, tail surfaces, tail rotor). A more detailed description of the airflow is obtained by measuring fuselage surface pressures and the velocity field near the aircraft. Not only average values are measured but also unsteady components [108,109,152,166]

An effort is also made to model the helicopter as accurately and completely as possible. The main and tail rotors are added to the fuselage, the engine airflow is simulated by installing small fans inside the model, and the engine and MGB oil cooling airflow can also be represented in the same way.

The purpose of this review is not to cover the wide range of wind tunnel tests in detail, and in fact this would be a difficult undertaking since many tests are conducted for design or development purposes and are not available for publication. The following sections will be devoted to an examination of general results obtained in the two major areas of fuselage aerodynamics namely drag and interactions.

2.1.3.1 Fuselage Drag

The first helicopter studies on fuselage streamlining were confronted with a new problem. Little attention had been given to the subject, since the traditional problems of vibration, stability and hover performance specific of the helicopter had mobilized all the engineering resources. The consequence of this lack of attention was a very high level of parasite drag and prospects of spectacular drag reductions; in [110] estimated that the equivalent flat plate drag area of a particular landing gear design could be reduced by as much as 20 ft^2 which represents about twice the total drag of a modern 4-ton helicopter. The situation has changed today, and the gains that can be expected are neither as significant nor as easy to obtain. Nevertheless the rising fuel costs since the first oil crisis in 1973 has prompted helicopter manufacturers to give very serious attention to parasite drag and to the means of reducing operating cost [111,112,113]. This has led to the establishment of an inventory of the available helicopter drag data. Reference [111], in particular, lists the points requiring special attention in order to ensure reasonable drag levels. Two of these points may be considered critical, in that a poor design can result in a catastrophic drag level: they are (1), the shape of the fuselage afterbody, and (2), the rotor head design.

Fuselage Afterbody Drag

This is the portion of the fuselage where flow separation is liable to occur producing a large increase of pressure drag [114]. The sharper the contraction, the stronger the adverse pressure gradients in the boundary layer at the rear of the fuselage causing the flow separation line to move upstream. In [95] it is shown how a minor modification of the shape of the aft fuselage involving a more gradual cross-sectional area contraction can produce a significant drag reduction.
Figure 71 shows the influence of the afterbody contraction ratio on fuselage drag. Since the drag coefficient of an aerodynamically "clean" helicopter fuselage (relative to its frontal area) is of the order of 0.1, it can be seen that a very sudden contraction can double the drag of the bare fuselage.
In [106], various fuselage models were tested with different afterbody longitudinal but also lateral contraction ratios. It was observed that both types of contraction produced similar drag increases, which is in agreement with [102].

2.1.2.2 Wind Tunnel Wall and Airstream Blockage Corrections

Reference [98] reviews the corrections required. In fuselage aerodynamics, when the rotor is not represented only the blocking effect must be considered. In the AEROSPATIALE Marigane wind tunnel (ϕ=3m test section), the model scale is chosen so as to enable the fuselage to be tested with the rotor (ϕ=1.5m); the fuselage dimensions are therefore small compared to those of the tunnel with a cross sectional area ratio of the order of 1:100. The blocking effect is minimal, and only minor corrections are required.

If now a portion of the helicopter is to be represented at large scale and the model occupies a significant fraction of the test section, then the airflow blockage effects cannot be neglected. References [97,99] report on tests of a 8:10 scale model of a rotor head made on the Hub Pylon Evaluation Rig (HPER) which is a mockup of the upper fuselage designed to measure the forces acting on the rotor head and pylon fairing. In this case the model occupied 10% of the test section. A comparison between surface pressures measured on this model and on a smaller model (3:10 scale) in the same wind tunnel showed that the blockage correction reached 14% of the dynamic pressure, which certainly cannot be neglected.

Wall effects must also be taken into account when the fuselage is tested with a rotor. Interactions between the wind tunnel airstream, the rotor wake and the tunnel walls impose a minimum tunnel speed below which the aerodynamic characteristics are biased.

2.1.2.3 Test Installation Effects

Helicopter fuselage models for wind tunnel testing are generally mounted on a single-mast device as shown in figure 70. The presence of the mast obviously perturbs the airflow around the aft portion of the fuselage; this can be extremely undesirable, in view of the fact that the afterbody section is a critical fuselage area, especially from the standpoint of aerodynamic drag. It is therefore absolutely necessary to evaluate the influence of the test installation on the measured aerodynamic characteristics.

One method of determining this influence is described in [100]. Two series of tests are conducted: one with the model alone mounted upside-down, the other with a mockup of the support structure. The difference between the corresponding tests in each series represents the perturbations due to the test installation, which may be quite substantial. The tests reported in [100] were intended to explain the differences observed between measurements made on a full scale aircraft in the NASA Ames wind tunnel and those made on a 1:5 scale model. It was concluded that these could be attributed for the most part to differences in the model support structure, while Reynolds number effects or geometrical modelling errors were much less significant.

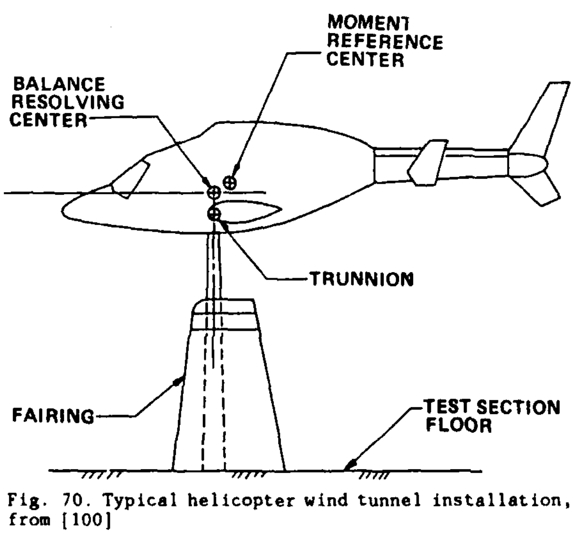

Fig. 70. Typical helicopter wind tunnel installation, from [100]

2.1.2.4 Fuselage Modelling Errors

When a helicopter is represented by a small scale model, two types of errors are introduced relative to the full scale aircraft geometry:

- intentional errors which are conciously introduced in order to simplify the construction of the model; details such as rivet heads, skin panel junctions, gaps between cowlings, etc. cannot be represented at small scale and are omited. A "clean" model is therefore tested, and drag corrections are introduced based on experimental results such as those of [101]. These "details" can account for a significant fraction of the drag: 14% of the total drag according to the estimation made in [102]

- unintentional errors arising from inaccuracies in building the model: care is obviously taken to minimize these errors by tightening the manufacturing tolerances.

Reference [100] contains an interesting study of the influence of modelling errors. A 1:5 scale model of a BELL-222 helicopter and a full scale prototype were tested in the wind tunnel, and large differences were recorded between the two series of tests, particularly with regard to drag. The geometrical characteristics of the two aircraft were carefully compared, and significant differences were found on the cowlings and fuselage. The 1:5 scale model was corrected and was then retested. Only minor differences were observed between the two series of tests on the small scale model. The drag of the corrected model was slightly lower but this was partly attributed to a better sealing around the rotor mast/fuselage junction. This result shows that the accuracy of the model is not very critical and that the models routinely used are of sufficient quality: geometrical modifications can be simulated with modelling clay to evaluate their influence on the aerodynamic characteristics of the fuselage without complicating the test procedure unnecessarily.

The influence of the model surface finish was also determined during the same study. Carborundum strips were bonded to the model to force the boundary layer transition and obtain turbulent flow as on the full scale aircraft. This change primarily affected the drag coefficient (+4% at zero incidence and sideslip), giving values closer to those measured at full scale.

Another factor to be considered when discussing the model similarity concerns the effect of internal airflows. The engine airflow and the MGB oil-cooling airflow can influence the aerodynamic characteristics of the aircraft. As a first approximation, the residual thrust from the exhaust nozzles can roughly be assumed to balance the intake momentum drag, so that the aerodynamic data measured on an inert model is approximately valid. However, this assumption turns out to be optimistic with regard to drag: full scale measurements [95] showed that the fuselage drag increased by approximately 4% when the engine was started.

Allowance can be made for this effect by making a simple momentum balance of the engine intake and exhaust airflows [102]. An unpublished AEROSPATIALE study showed that this method gave good results on a 1:4 scale model of the AS 355 TWINSTAR helicopter simulating the engine flows.

These airflows can also be simulated in the wind tunnel, although not without difficulty. Because of the temperature difference between the engine air intake and exhaust gases, different flowrates are required to simulate the airflow near the air intake and the exhaust nozzle. This requires either two series of tests or an additional air input into the model via the support mast, with all the interference problems with the balance that this solution entails.

The most effective means of studying the drag reductions possible on internal airflows is with full scale wind tunnel tests, such as those conducted between 1935 and 1945 on World War II fighters and light bombers which led to reductions in total drag approaching 25% in some cases [103].

2.1.2 Wind Tunnel Testing

Compared with flight tests, wind tunnel tests offer a number of definite advantages [90]:

- lower cost
- measurements can easily be made directly on the model as well as in the surrounding airflow
- the model scale can be adapted to the test objectives: from small scale models for studying general aircraft configurations, to full scale models for measuring the drag induced by protruding components (antennas, door handles, footsteps, etc.)
- dangerous flight conditions can be simulated
- the effect of individual aircraft components can be isolated.

Conversely, problems arise in several specific areas resulting in deviations from true flight test conditions [92]:

- Reynolds number effects
- wind tunnel wall and airstream blockage effects
- interference due to model support structures
- geometrical inaccuracies in the model itself.

2.1.2.1 Mach and Reynolds Number Effects

Airflow similarity requires that the Mach and Reynolds numbers be accurately reproduced in the wind tunnel tests. In helicopter fuselage aerodynamics, the forward airspeed is always well below the speed of sound, and, as the flow is practically incompressible, Mach number similitude need not be rigourous. Only high speed helicopter tests show a slight fuselage drag increase above Mach 0.4 (figure 68).

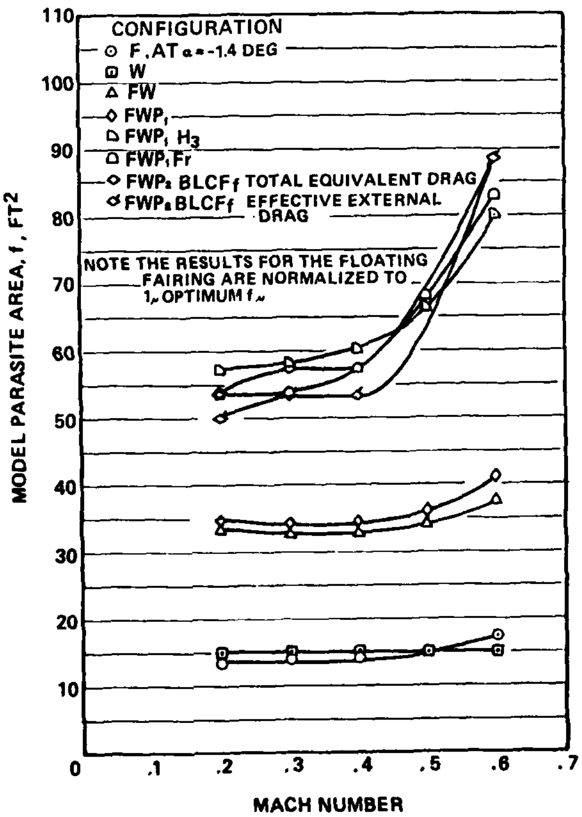

Fig. 68. Mach number effect on helicopter drag (Fuselage, Wing, Pylon, Hub, Boundary Layer Control, Floating Fairing), from [93]

Figure 68 also shows the effect of Mach number on rotor head drag, which also increases above Mach 0.4 but is much more sensitive. However this only concerns helicopters flying at airspeeds exceeding 270 knots at sea level, so that the Mach number effect can be disregarded for conventional helicopter designs.

Mach number similitude is thus not a problem. However, large differences arise in the Reynolds number between a flight test and a wind tunnel test on a small scale model which can result in significant discrepancies with regard to drag.

The effect of Reynolds number was evaluated in tests described in reference [94]. For values ranging from 2.4×10^6 and 17.5×10^6 per meter, few differences are noted on the fuselage pitching moment and lift curves.

Figure 69 shows that the Reynolds number has only a limited effect on fuselage and cowling drag, matching the estimated variations in skin friction drag. This result is in agreement with earlier investigations reported in [95,96]. On the other hand, in reference [97], fuselage drag was found to decrease at low Reynolds numbers. This surprising result was attributed to an interaction occuring at low airspeeds between the wind tunnel airstream and the engine airflow which was also simulated. The drag of the rotor mast fairing was also measured in [97] but at lower Reynolds numbers than in [94] (3×10^4 to 4×10^5 per meter) and a critical Reynolds number was determined: above 10^5 per meter the pylon fairing drag remains constant while it rises quickly as the Reynolds number decreases below this value.

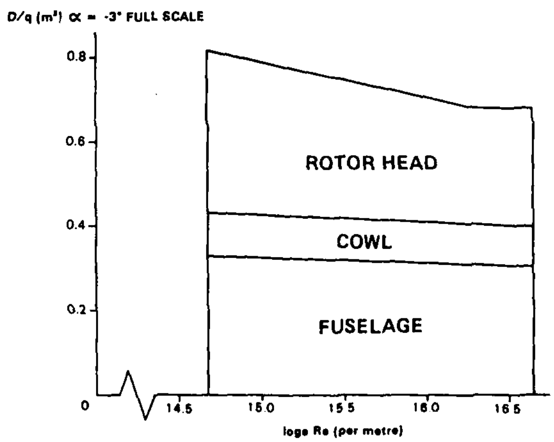

Fig. 69. Reynolds number effect on helicopter drag, from [94]

Turning to rotor head drag, figure 69 shows that the Reynolds number effect is no longer negligible. Reference [97] describes a similar effect with a critical Reynolds number of the same order of magnitude as for the pylon fairing drag, and attributes this variation to the transition from a subcritical to a supercritical Reynolds number relative to the rotor head component size. In reference [96], the Reynolds number was found to have little effect on the drag of faired rotor hubs. For unfaired hubs, however, the drag was found to increase with the Reynolds number, but SHEEHY suggests that this discrepancy is due to poor geometrical representation of the rotor head on the small scale model.

The problem is thus less clear for the Reynolds number than for the Mach number. Nevertheless, the Reynolds number appears to have little effect on fuselage drag if care is taken to avoid testing very small models at very low airspeeds. The effect on rotor head drag seems more important, and the Reynolds numbers should be bracketed for purposes of comparison.

2.0 FUSELAGE AERODYNAMICS

"Helicopter aerodynamics" have long been synonymous with "rotor aerodynamics", and only in recent years have helicopter designers been much concerned by the aerodynamics of the fuselage. Fuel efficiency requirements following successive oil crises and the higher airspeeds that are now expected of helicopters have highlighted the need for serious attention to the aerodynamics of the fuselage and to its impact on aircraft drag and flying qualities. As long as helicopters were primarily considered as hovering aircraft, the fuselage could be reduced to the barest minimum: a canopy and a trussed tail structure (figure 65). With forward airspeeds approaching 300 km/h, streamlining has become critical (figure 66), and even more sophisticated aerodynamic solutions will be required for the helicopters now on the drawing boards capable of flying at 200 knots (figure 67).

Fig. 65. SO 1221 DJINN

Analytical methods, discussed in the next section, are currently unable to deal with the most critical problems raised by fuselage aerodynamics. The existence of large zones of separated flow at the rear of the fuselage, the presence of a rotor head that is not only geometrically complex but also rotating, the downwash induced by the main rotor are just a few of the phenomena which present serious modelling difficulties and for which no satisfactory analytical solutions are available today. For obvious reasons flight tests are generally only conducted in the final stages of the development phase after extensive wind tunnel testing which remains the most appropriate and widely used design tool in spite of some specific problems which will be considered later in detail. Experimental data on fuselage drag is then reviewed before discussing the effect of fuselage and rotor hub wake interference with the tail surfaces.

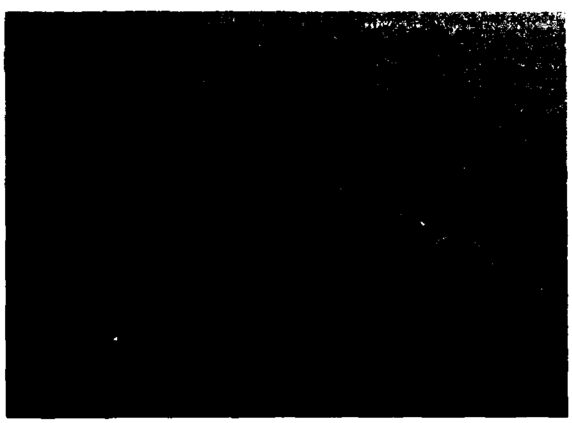

Fig. 66. SA 365N DOLPHIN

Fig. 67. HUGHES LHX Design

2.1 EXPERIMENTAL METHODS

Two broad categories of methods are used in experimental fuselage aerodynamics: flight tests and wind tunnel tests (or hydrodynamic tunnel tests, but these are seldom used). Both methods range from simple qualitative flow visualization tests to detailed measurements of the pressure and velocity fields which offer an accurate description of the fuselage airflow.

2.1.1 Flight Testing

A helicopter is built to fly, and from this standpoint a flight test under true operating conditions is irreplaceable. The test instrumentation must simply be set up so that it does not alter the aerodynamic configuration of the aircraft. Despite this unquestionable advantage, flight tests are nevertheless seldom used for basic research in fuselage aerodynamics.

Drag, for example, is one of the most important fuselage characteristics since it determines helicopter performance to a large extent (fuel consumption, top speed, etc.), and yet it is not directly measured in flight. Instead, drag must be related to another measurable parameter such as the rotor shaft power [90] or the level flight speed achieved at a given collective pitch setting from which it has to be extracted by comparison with a reference configuration. This is not always easy to achieve however. For example a significant reduction of the parasite drag of the aircraft will result in a change of attitude of the helicopter in flight and the measured rotor shaft power variation will then not only reflect the direct effect of streamlining but also the effect of the aerodynamic angle of attack change on fuselage parasite drag. Flight tests do not offer the possibility to isolate all the relevant factors, nor do they give access to absolute measurements of such essential parameters as aircraft drag or other aerodynamic coefficients but only allow comparisons to be made between similar configurations under imposed flight conditions.

NASA's Rotor Systems Research Aircraft (RSRA) [91] which is equipped with multi-balance systems between various aircraft components does permit direct in-flight measurements of the aerodynamic forces and moments acting on several structural elements. However, this aircraft was designed to serve more as a rotor test rig than as a fuselage research tool.

In any event, the problems related to flight testing are not only measurement problems. The cost factor must be considered, and this alone may explain why flight tests are little used in fuselage aerodynamics. While it is conceivable to flight test a new blade design despite the cost incurred, it is out of the question to design a new fuselage, i.e. a new aircraft, simply to evaluate its aerodynamic characteristics. Only very limited fuselage shape modifications can in fact be tested in flight. Flight tests thus constitute a valuable development tool, but are of little practical use in project studies or basic research, where the wind tunnel plays a fundamental role.

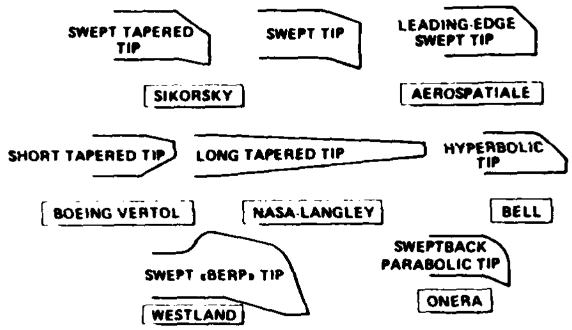

Fig. 62. Typical blade tip designs, from [5]

The results obtained by SIKORSKY, BOEING-VERTOL, MBB and ONERA with sweptback and trapezoidal tips have already been discussed. In §1.2.2. we mentioned some of the problems encountered with sweptback tips. The development of steady and unsteady 3-D flow analysis methods (c.f. §1.1.6.) will allow blade designers to define better tip shapes able to meet specified aerodynamic criteria. As an example, the iso-Mach lines computed at $\psi=90^\circ$ (figure 63) clearly show that a simple 45° backward sweep of the blade leading edge, or a more complex shape such as the parabolic sweptback tip, are effective in attenuating the shock-waves on the advancing blade in high-speed flight thereby reducing the rotor power requirement and the quadripolar impulsive noise generated.

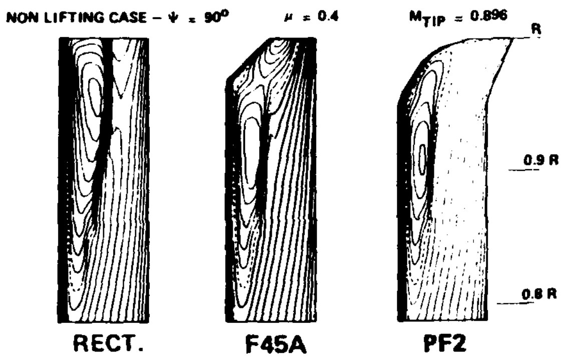

Fig. 63. Typical Iso-Mach lines (ONERA code, NACA 0009 blade, λ=15), from [5]

A sweptback tip may also produce blade torsional deformations which can reduce the negative lift forces appearing at the tip of the advancing blade in high-speed flight, and also delay the onset of retreating blade stall. By using blades with relatively high torsional flexibility and with suitably tuned natural frequencies it may be possible to design rotors with exceptional performance characteristics. This is the concept behind the aerodynamically conformable rotor on which SIKORSKY [87,88] and NASA [89] have been working for a number of years.

The difficulties of correctly mastering the problems involved are well illustrated by comparing the very promising test results obtained by SIKORSKY [88] with rotors having a large elastic twist activity ensuring nose-up twisting on the advancing blade and those of NASA [89] which came to a quite different conclusion, the best rotors tested being those experiencing the smallest torsional deformation.

This underlines the absolute necessity of developing rigorous coupled aeroelastic codes with 3-D unsteady aerodynamics to verify that the desired blade deformations actually occur in the desired azimuthal sector.

The problems of blade torsional deformation are theoretically minimized with straight trapezoidal blades, for which the reduction of the rotor power requirement is obtained by chord tapering of the blade tip sections (where the aerodynamic drag is the highest). However, it is necessary to compensate for the loss of lift at the blade tip by an increase of chord inboard to achieve the same load factor capability as the basic rectangular blade. A simple rule is to select a spanwise chord distribution l(x) such that $\int^1 l(x)x^2dx$ remains constant (i.e. equal to the equivalent rectangular blade value). When applied to a trapezoidal blade the quadratic chord weighting rule results in a net increase of blade area, and consequently of blade weight, which offsets some of the performance benefits.

Figure 64 also shows that the shock wave is probably at least as strong on straight trapezoidal tips as on rectangular tips: this observation leaves little hope of attenuating the high-speed impulsive noise generated by these shock waves.

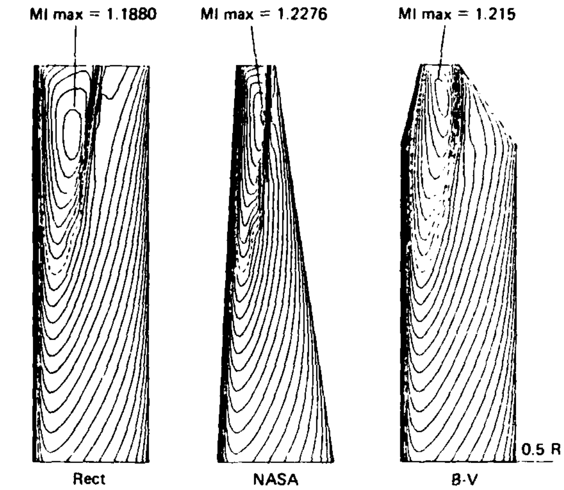

Fig. 64. Effect of taper on the flow around non-lifting rotor blade tips (ONERA code, ψ=90°), from [5]

A satisfactory compromise at the present time might consist in using blade tips with the leading edge swept back progressively to relatively high angles in combination with significant chord taper (e.g. 1:3) and to keep the trailing edge straight in order to minimize the control loads. It should be interesting in this respect to follow the tests to be carried out with the BERP tip that WESTLAND plans to install on the LYNX-3 and EH-101 helicopters and which is expected to alleviate both retreating blade stall and advancing blade compressibility problems.

1.4 CONCLUSIONS ON ROTOR AERODYNAMICS

We have seen that rotor aerodynamics sets numerous and complicated problems both in hover and forward flight. If experimental methods are able to describe quite well the rotor operating conditions, the theoretical methods are still far from being capable of offering accurate predictions of the local instantaneous aerodynamic loads on the blades. If we want to make significant progress in this area, it is necessary to:

- develop lifting-surface codes for unsteady 3-D flows in forward flight and quasi-steady flow in hover to calculate realistic pressure distributions in the presence of a free wake.

- start aerodynamic drag calculations from pressure distributions using a viscous-inviscid fluid coupling technique.

- estimate in the most consistent manner the blade deformations (particularily the elastic twist) with the hope that fully coupled aeroelastic codes will become available in the near future.

To check the validity of these models, it is essential to conduct well instrumented wind tunnel and flight tests with numerous pressure and stress measurements as well as direct measurements of the blade deformations so as to verify each step of the calculation separately.

1.3.2 Blade Twist

For pure hover performance it is recommended to use blade twist values of at least -12° to -14° which provide a more uniform induced velocity field through the rotor while ensuring that the blade tip sections operate below their drag divergence Mach number. Figure 58, from [84], shows that by increasing blade twist from -8° to -14° a 3 % power saving was achieved at a C_T/σ=0.11 and a tip speed of 196 m/s (Mtip=0.575); at 226 m/s (Mtip=0.663) the corresponding reduction was 8%. It may be noted that the figure of merit of a rotor with highly twisted blades is practically always higher than that of a rotor with small blade twist regardless of the thrust level.

It is known that the tip vortex strongly affects the spanwise lift distribution, especially near the blade tip, increasing the angle of attack outboard and reducing it inboard. A non-linear twist distribution can be used to attenuate the detrimental effect of the vortex interaction by smoothing out the angle of attack variation along the blade span. Figure 59 from [85], shows that the maximum rotor figure of merit is significantly higher (approx. 5%) and that the C_T/σ at which maximum figure of merit is achieved is increased by 0.02. This non-linear twist distribution was defined by AEROSPATIALE using the method described in [44]. Tests demonstrated however that a 2% power penalty was incurred in high speed forward flight.

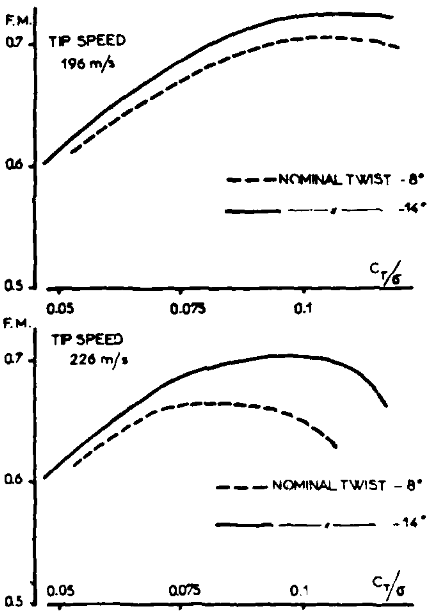

Fig. 58. Effect of blade twist on rotor figure of merit (ONERA S2 Chalais WTT), from [84]

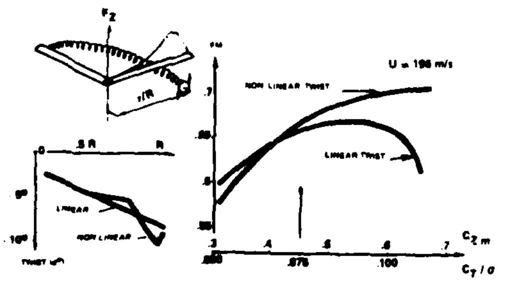

Fig. 59. Effect of blade non-linear twist on rotor hover performance (AEROSPATIALE design, ONERA tests), from [85]

To improve still further the rotor hover performance it is possible to combine large blade twist with sweptback blade tips, with anhedral. Tests made by SIKORSY on model rotors of the BLACK HAWK (with -16° equivalent linear twist) and of the S-76 (-10° twist) showed that the addition of 20° of anhedral to the swept-tapered blade tips improved the rotor figure of merit by approximately 3% in both cases (figures 60 and 61 from [86]). This can probably be attributed to the fact that the anhedral tip changes the maximum blade circulation and the tip vortex position relative to the following blade.

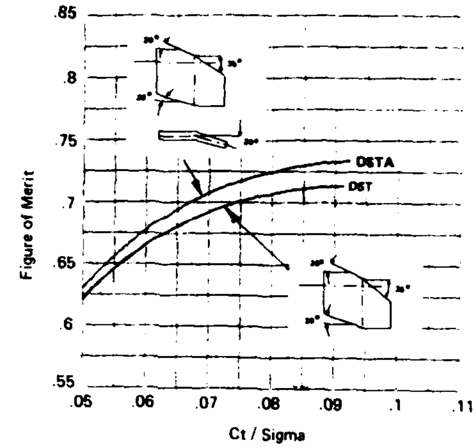

Fig. 60. Effect of blade tip anhedral on BLACK-HAWK hover performance, Mtip=0.6, from [86]

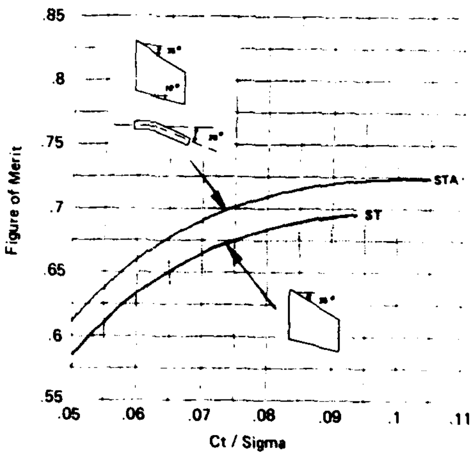

Fig. 61. Effect of blade tip anhedral on S-76 hover performance, Mtip=0.6, from [86]

1.3.3 Blade planform

The blade tips are of considerable importance since, (1), they sustain the highest dynamic pressures, (2), they are at the origin of the formation of the tip vortices, and (3), they generate most of the rotor drag and noise.

Figure 62 shows some of the tip shapes already in production or presently under development. This subject is covered in detail in [5], and we do not intend to review here the merits and problems of each design.

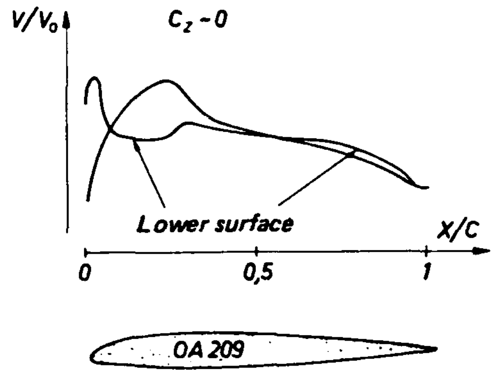

Fig. 54. Design of OA-209 airfoil, from [78]

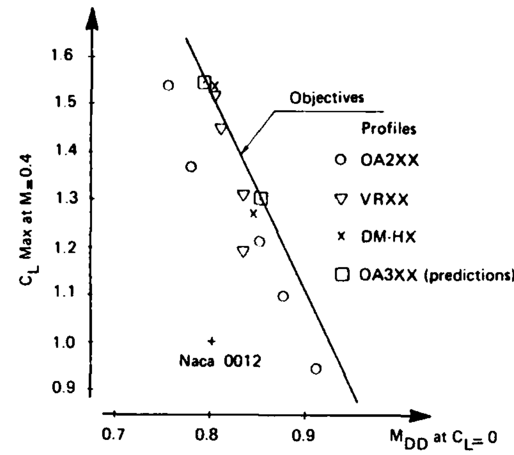

Fig. 56. Main aerodynamic characteristics of new airfoils for helicopter blades

The other airfoils were defined with direct methods by geometrical transformation of the OA209 with the exception of the OA213 which was again defined by an inverse method, specifying a velocity distribution at M=0.5 and Cl=1, as shown in figure 55. The next step consisted in estimating the airfoil performance characteristics by means of a program that solves the full potential equation using a non-conservative scheme with weak boundary layer coupling [80], comparable to the classical GARABEDIAN & KORN codes. The predicted airfoil performance is then checked by wind tunnel tests. Figure 56 summarizes the main performance characteristics of the OA2XX airfoils which are shown here compared with the VRXX airfoils developed by BOEING-VERTOL [54,81] and with the more recent DM-HX airfoil series designed by the DFVLR and MBB [82].

One can measure here the progress made in the design of advanced airfoils since the time when the old NACA 0012 used to equip so many helicopter rotor blades. The same figure also indicates AEROSPATIALE's design objectives: it can be seen that the new performance goals concern airfoils whose relative thickness is between 9% and 13% with a high Clmax at Mach 0.4 associated with a low Cmo to avoid high control loads, and with the highest possible drag divergence Mach number at near zero Cl.

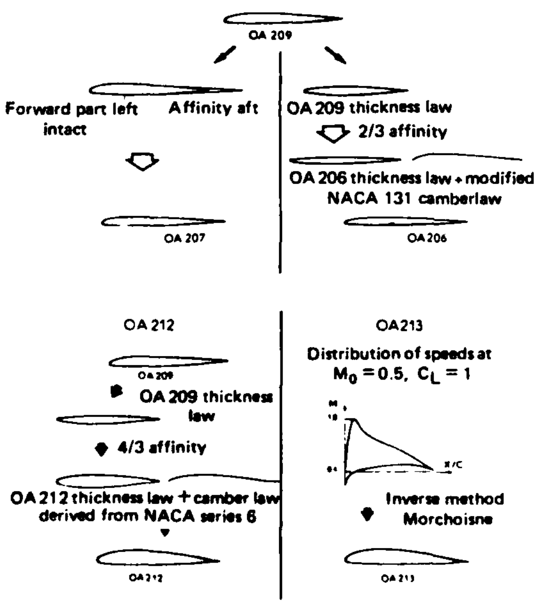

Fig. 55. Design methodology of OA-207, OA-206, OA-212 and OA-213 airfoils, from [78]

It is in fact questionable that thin airfoil sections at the tip of the blades should be better in high speed forward flight as AEROSPATIALE rotor tests in the MODANE S1 wind tunnel have shown. Figure 57 gives the definition of the rotors tested. Figure 2 clearly shows that a thick airfoil over most of the blade is preferable (rotor 7A versus rotor 6B), and that the constant 9% thick airfoil section at the tip was more efficient at high speed than the tip tapered from 9% to 6% which is in contradiction with the theoretical predictions. It is recalled that the drag divergence Mach number of the OA206 airfoil is 0.91 compared with 0.89 for the OA207 and 0.85 for the OA209. A satisfactory explanation of these results will probably be possible only when 3-D unsteady flow methods are developed.

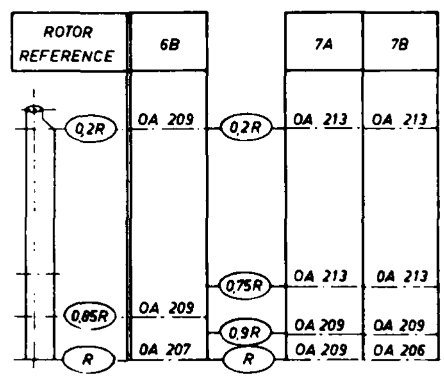

Fig. 57. AEROSPATIALE model rotor blade definition for S1 Modane WTT, from [5]

New airfoil numerical optimization techniques presently in development appear very promising as a design optimization tool to elaborate a satisfactory compromise between the low-speed high Cl and high-speed low Cd performance requirements. ONERA has applied such techniques to the optimization of the OA207 airfoil [83] and achieved a 16 % reduction in Cdo at M=0.85 and a 6% reduction in Cd at M=0.4 and Cl=0.83. ONERA is now working on the design of a new airfoil family (OA3XX) with computed Clmax and Mdd performance at least equivalent to the VRXX and DM-HX airfoils (figure 56).

In concluding this section on flight tests, the comments already made (§ 1.2.1) concerning the difficulty of estimating the exact rotor forces in flight remain applicable here. Tests to be conducted in the USA on the RSRA research aircraft appear very promising in this respect since the rotor will in fact be mounted on a calibrated balance, as illustrated on figure 52 from [75]. The auxiliary propulsion system and wings make it possible to change the rotor lift and propulsive force as desired, making this vehicle potentially a "flying wind tunnel".

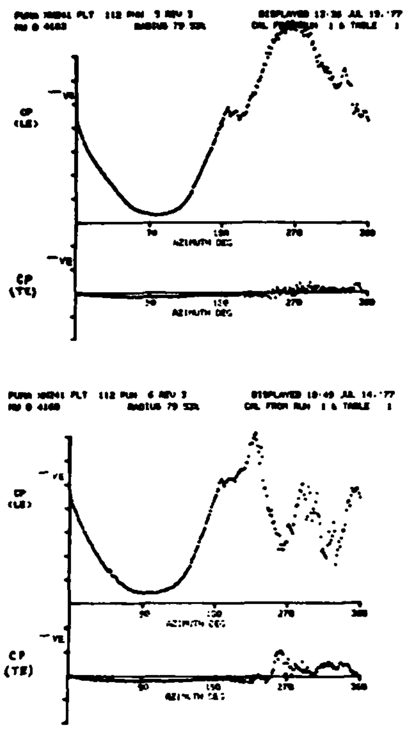

Fig. 51. L.E. and T.E. pressures, (a) before, (b) after stall, as measured in flight by RAE, from [72]

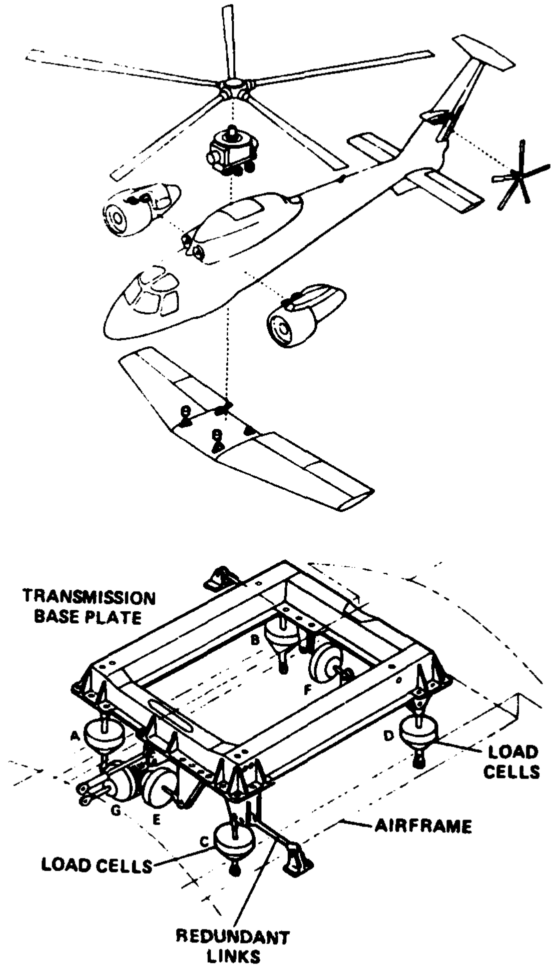

Fig. 52. Rotor force and moment measurement system on RSRA aircraft, from [75]

1.3 ROTOR DESIGN

This heading covers the geometrical definition of the rotor: diameter, number of blades, chord, airfoil distribution, blade twist and planform which may include a complex tip design. Blade airfoil selection remains a major aspect of the problem. It is still based on 2-D steady performance, which is the only one that can be effectively computed at present. The following discussion is limited to airfoil selection criteria, blade twist, planform and tip shape.

1.3.1 Airfoil Selection

The choice of airfoils is determined by the helicopter operational requirements as stated in the general specifications. The airfoil operating conditions are very much dependent on the flight configuration so that it is impossible to define a single optimal airfoil and that a tradeoff between conflicting requirements is always necessary.

In practice, depending on the blade spanwise section considered, the goal is to balance the advancing blade airfoil requirements (high drag divergence Mach number at small lift coefficients) with those of the retreating blade (high Clmax at low Mach numbers) while maintaining a good lift/drag ratio at the intermediate values of lift coefficient and Mach number on the fore and aft blades and in hover.

Figure 53 summarizes the specifications set by AEROSPATIALE to ONERA several years ago to design a set of helicopter airfoils. References [76,77] provide ample information on ONERA's design methodology. Figure 54 shows that the basic airfoil of the OA family, the OA209 airfoil, was defined by an inverse method [74] by specifying a velocity distribution at low Mach number and near zero lift.

FLIGHT CONDITIONS	PREPONDERANT AERODYNAMIC COEFFICIENT	SECTIONS				
		1		2	3	
FORWARD FLIGHT	Mdd at CL = 0 ≥	0.75	0.80	0.85	0.90	0.92
	$\lvert C_{m_0} \rvert$ ≤	0.01	0.01	0.01	0.01	0.01
HOVERING	L/D RATIO AT M_0 0.5-0.6 CL = 0.6 ≥	80	75	80	85	85
MANEUVER	$C_{L\,MAX}$ M_0 = 0.3 ≥		1.5	1.4		
	$C_{L\,MAX}$ M_0 = 0.4 ≥	1.6		1.3	1.0	0.95
	$C_{L\,MAX}$ M_0 = 0.5 ≥		1.3			
GEOMETRICAL CONSTRAINT	t/c	13	12	9	7	6

Fig. 53. AEROSPATIALE specifications for airfoil design, from [76]

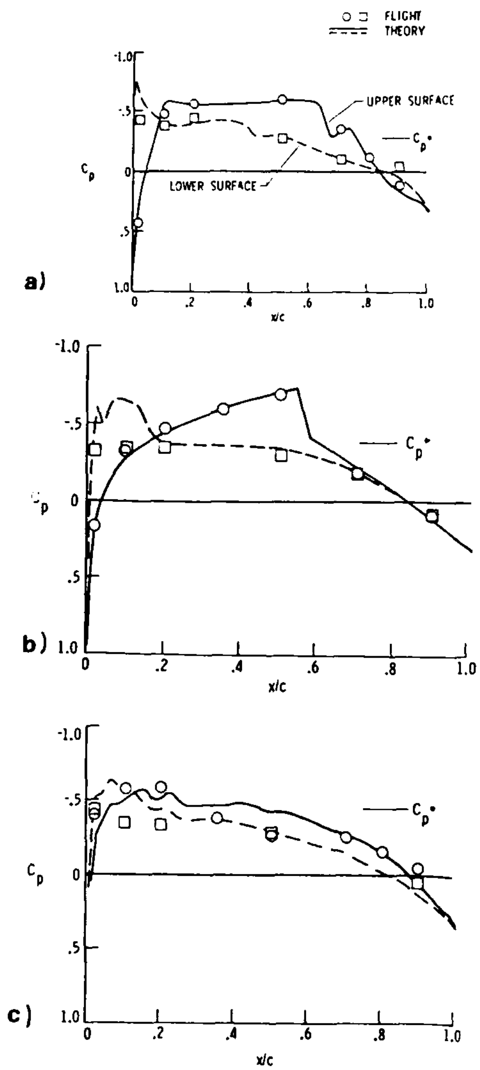

Fig. 47. Airfoil pressure distributions measured in flight at 0.9R and ψ=90° (NASA flight tests), from [68]

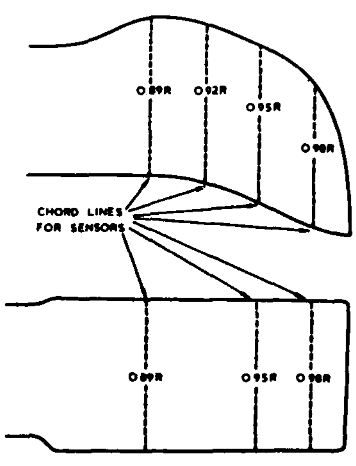

Fig. 48. RAE tests on PUMA blade tips, from [70]

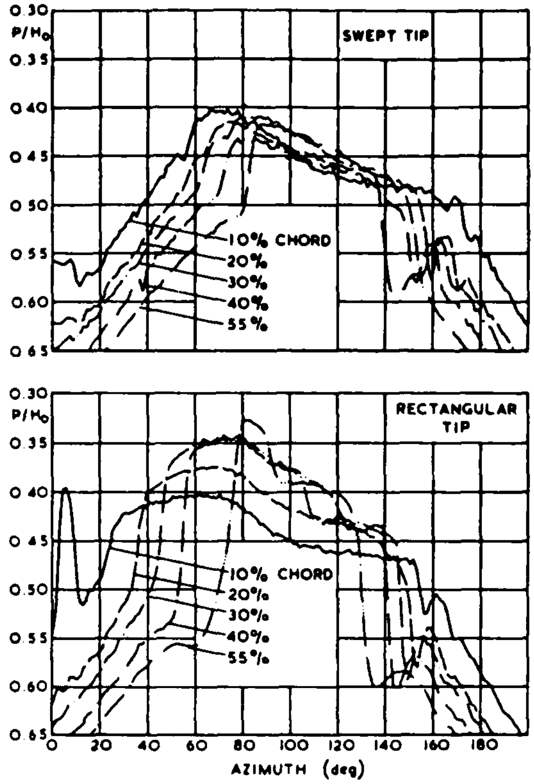

Fig. 49. Pressure measurements on PUMA blade tips at 0.95R (RAE tests), from [70]

Still another objective may be to acquire aerodynamic data over the entire blade span. Ideally, the blade should be equipped with numerous measurement sections distributed along the full span. In practice, however, it is difficult and very expensive to install hundreds of sensors for a single experiment. An intermediate solution is to install two sensors per section, both on the upper surface, one very close to the leading edge, and the other close to the trailing edge. Figure 50 clearly indicates that the normal force coefficient and the pressure at 0.02c from the leading edge show very similar azimuthal behavior. Figure 51 shows that the sensor at 0.91c can be used as a stall indicator. This method which has been in use for many years at RAE proves to be extremely valuable whenever it has been possible to correlate a local pressure measurement (Cp) with a global airfoil parameter (such as CN or α). Such correlations are naturally established on an empirical basis; while these are conceivable in 2-D steady flow, it is most unlikely that they remain valid in 3-D unsteady flow. The RAE has nevetheless done considerable research in this area, analyzing oscillating airfoil data and the effect of dynamic stall [73,74], which unquestionably helps to understand the flow around a helicopter blade.

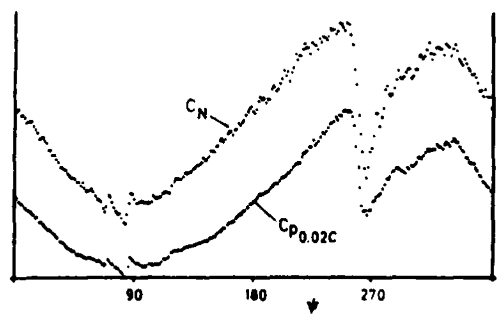

Fig. 50. Correlation between normal force coefficient and L.E. pressure as measured in flight by RAE, from [71]

AEROSPATIALE recently tested a 365N "DOLPHIN" with blades having sweptback parabolic tips defined by ONERA [35]. Here again, blade control load penalties were incurred, but power savings were recorded both in hover and in forward flight as indicated by figures 44 and 45. The result was a 2% increase in takeoff weight, a reduction in the rotor power requirement between 1% and 6% depending on airspeed and gross weight and a noise reduction of over 1 EPN dB.

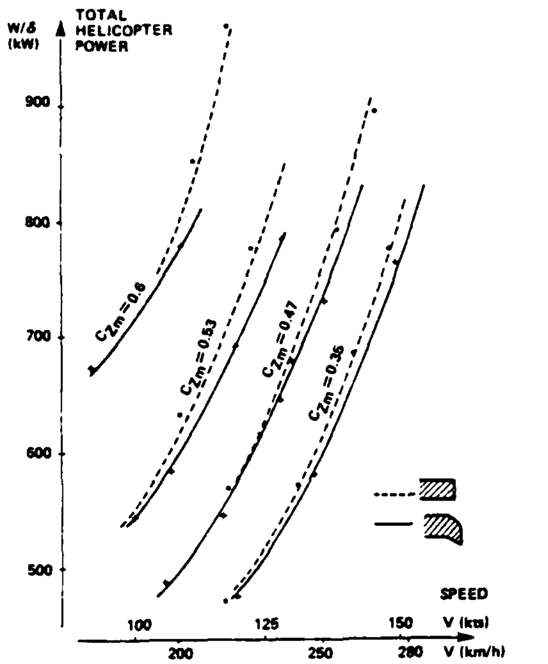

Fig. 44. AS-365N performance improvement in forward flight with parabolic sweptback tips, from [35]

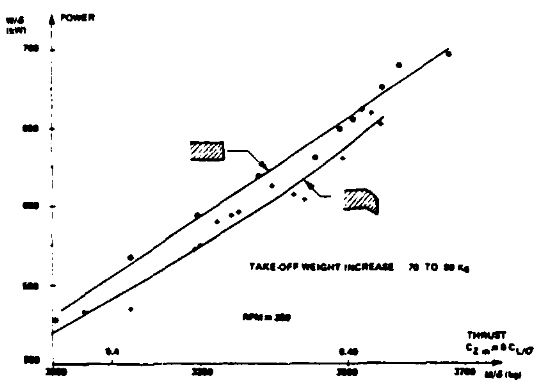

Fig. 45. AS-365N performance improvement in hover with parabolic sweptback tips, from [35]

1.2.2.2 Blade Pressure Measurements

Blade pressure measurements may have several objectives. One is to assess the local aerodynamic behavior of the airfoil sections. For example NASA equipped a standard BELL AH-1G rotor with 5 measurement sections . A gloved blade technique was adopted to install the pressure sensors without affecting the structural integrity of the blades, with only minor modifications to the airfoil [65]. A computerized data processing system was developed to analyze the enormous amount of data generated by these tests.The DATA-MAP software (Data from Aeromechanics Test and Analysis-Management and Analysis Package) allows comparisons between computed and test results to be made easily. Figure 46 compares measured iso-normal-force lines with those computed with the C-81 Rotor Flight Simulation Program described in reference [67].

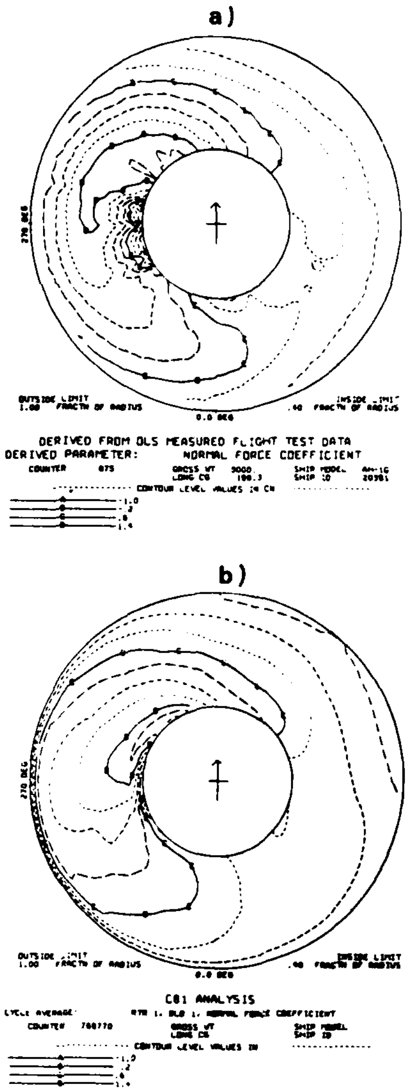

Fig. 46. Contour of main rotor normal force coefficients, (a) flight test data, (b) C81 analysis, from [66]

Another objective may be to evaluate new airfoil designs. NASA tested on the same aircraft three sets of blades with different airfoils (NLR-1T, 010-64C and 1C-SC2) and a single measurement section at 0,9 R equipped with 14 pressure transducers. The first results were published in [68]. Figure 47 shows a typical pressure distribution at $\psi=90°$ compared with the calculated 2-D pressure distributions at the same Mach number and lift coefficient as in the flight test. Many years ago the RAE tested simultaneously in hover two airfoils (NACA 0012 and RAE 9615) on a WESSEX helicopter on opposite blades at the same radial position [69]. This procedure permits direct comparisons to be made between airfoils under identical flight condition.

A third objective is to verify a new blade tip design. The RAE more recently equipped an AEROSPATIALE "PUMA" with 2 opposite blades having, one, a rectangular tip, and the other a sweptback tip. Figures 48 and 49 from reference [70] show these heavily instrumented blade tips together with typical pressure signals. The pressure time history at 0,95 R clearly shows that the sweptback tip delays the onset of shock waves in the advancing blade sector and strongly attenuates the recompression in a large azimuthal sector. This type of data is absolutely necessary to understand the performance of new blade designs and to validate analytical methods.

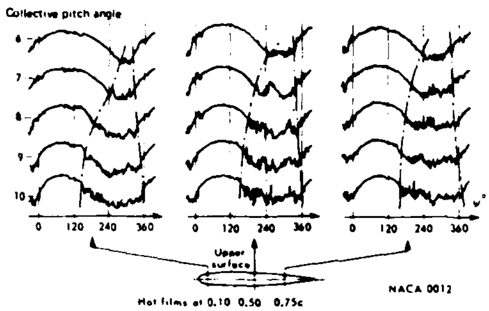

Fig. 39. Flow separation on rotor blades (ONERA tests with hot film detectors), from [57]

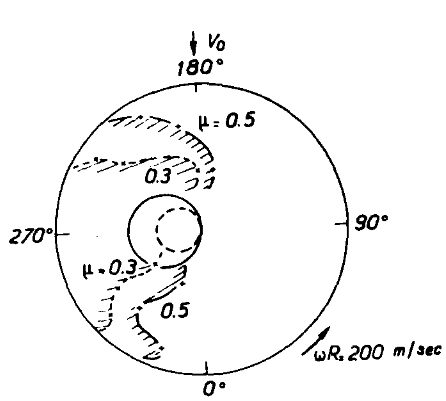

Fig. 40. Separation regions on the rotor disk as a function of advance ratio (CT/σ=0.085), from [57]

coming into widespread use and are extensively used by research organisations such as NASA [59] and IMFM [60,61] in France. They are gradually replacing hot wire probes and 2-D or 3-D pressure probes. Figure 41 shows the distribution of circulation measured on two-bladed rotors with straight and "ogee" tips.

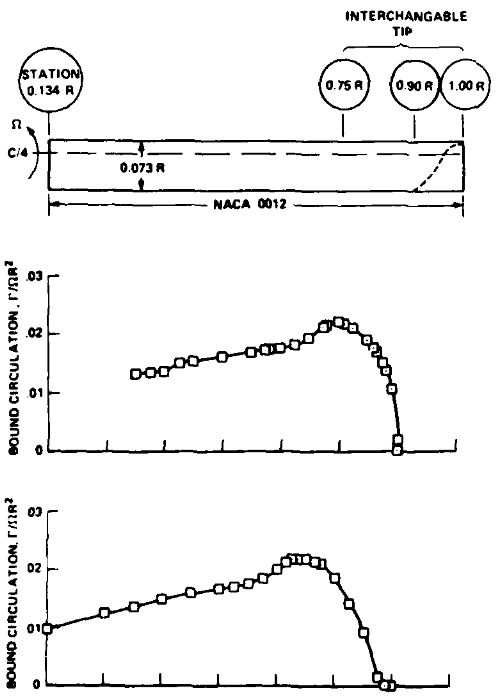

Fig. 41. Measured bound circulation on a 2-bladed rotor, (a) square, (b) ogee blade tip, from [59]

Laser velocimeters are also used in forward flight to determine the tip vortex trajectories, wake distortions and rotor induced velocities. UTRC has made a detailed study of the rotor wake geometry at small advance ratios [62] and ONERA has started to investigate the induced velocity field at high speed [63]. These measurements will be used to evaluate various methods for calculating the unsteady induced velocities and tip vortex trajectories in forward flight.

1.2.2 Flight Tests

Flight tests are of great importance since they are the ultimate step in the design process and reflect the success or failure of the aircraft. To the manufacturer, they must provide all the data necessary to confirm the aircraft design analyses. Flight tests thus reveal the weaknesses and deficiencies of the design methods. Finally, from a research standpoint, they provide a means for quickly verifying the validity of a new concept.

1.2.2.1 Global Performance Measurements

These tests are conducted to measure the helicopter power requirement as well as the vibrations and blade stress levels, and to determine the flight envelope of the aircraft (maximum airspeed, load factor, service ceiling, etc.).

For example figures 42 and 43 concern advanced blade tips tested by MBB. The AGB-IV tip significantly reduced the power required in forward flight but at the cost of a very large increase of the static control loads with respect to the conventional rectangular blade design.

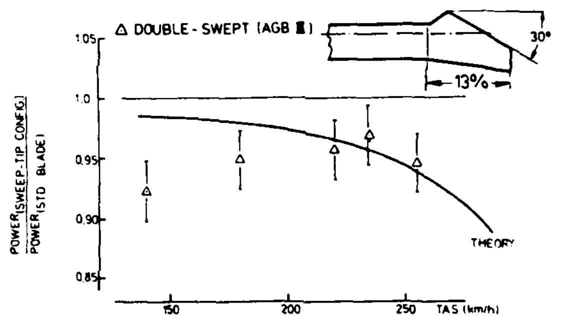

Fig. 42. Power saving obtained with a double swept tip on BO-105, from [64]

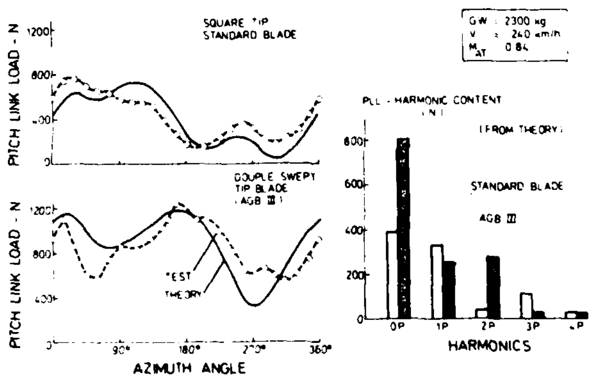

Fig. 43. Control loads for square and double swept tips on the BO-105, from [64]

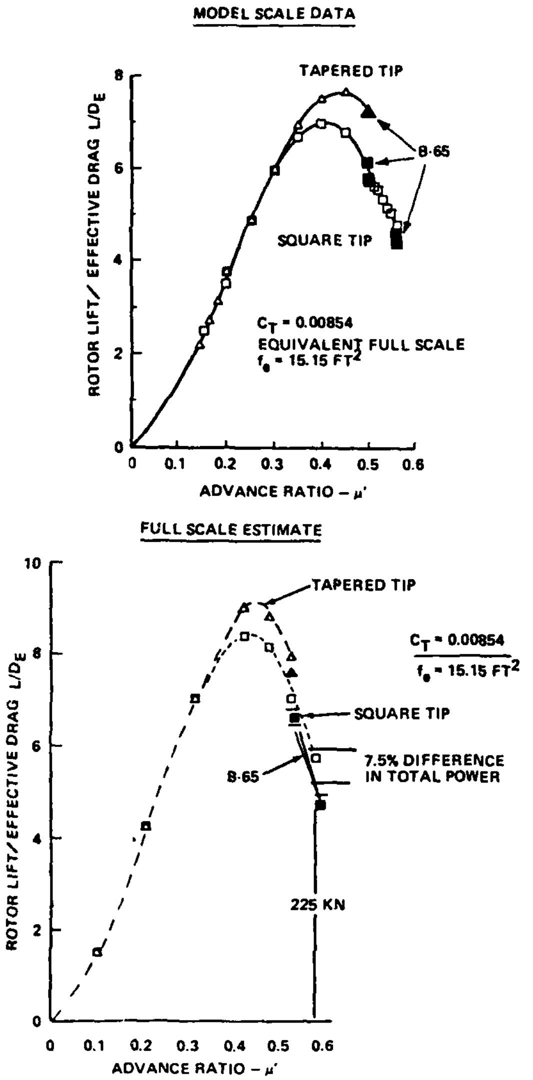

Fig. 36. Comparison of BOEING-VERTOL theory with model test data and full scale estimates, from [53]

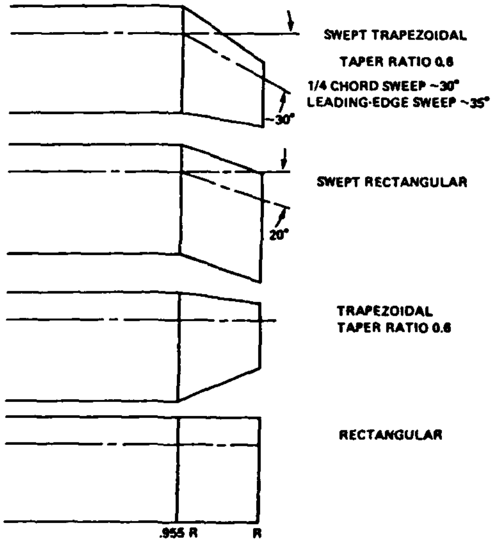

Fig. 37. Blade tip shapes tested by SIKORSKY in the 40x80ft Ames wind tunnel, from [55]

SIKORSKY tested four different blade tips on a full scale rotor, and found that the best results, particularily at high speed, were obtained with a swept trapezoidal tip having a taper ratio of 0.6 (selected on the S-76 rotor) as shown in figures 37 and 38 although the straight trapezoidal tip gave very similar results in this case.

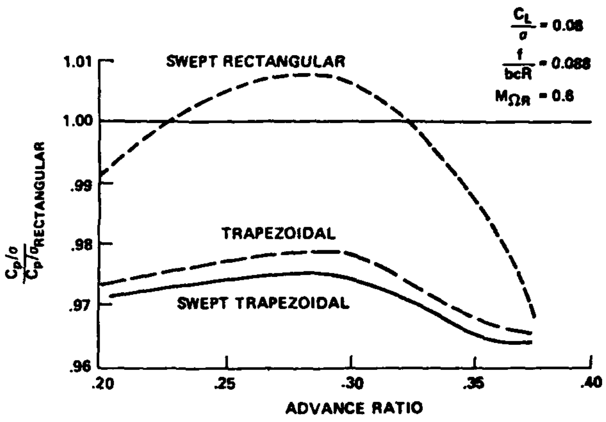

Fig. 38. Effect of tip shape on rotor power (SIKORSKY tests), from [55]

1.2.1.2 Local Measurements

Total force, moment and torque measurements generally provide insufficient information to understand rotor performance, and local measurements are of fundamental importance.

Strain gauges on the blades and pitch control rods provide a more refined analysis of the blade loads as a function of azimuth position. If strain gauges are installed in sufficient quantity and if the blade modes are properly identified in rotation, a relatively precise description of the blade deformations can be achieved. The RAE recently developed a Strain Pattern Analysis method that gives good results on small scale models [56] and will soon be applied to blades in flight. Blade strain measurements are essential to the qualification of aeroelastic prediction codes.

From an aerodynamic standpoint, however, blade pressure measurements are irreplaceable since they can be used to determine the instantaneous airfoil lift and pitching moment but can also detect the presence of shock waves or major flow separation. Such measurements are nevertheless difficult to make and their accuracy, particularily as regards pitching moment, is generally restricted by the limited number of pressure transducers that can be installed in a small blade section. Pressure measurements such as those made by ONERA or US Army RTL (figures 15, 16 and 18) clearly show the usefulness of such data in the analysis of rotor operation and the validation of theoretical methods.

Hot film detectors (which are easier to install than pressure transducers) give interesting information on the state of the boundary layer particularily as regards blade stall. Figures 39 and 40 show typical hot film signals and the corresponding separation zones determined on AEROSPATIALE rotors tested in the ONERA S1 wind tunnel at MODANE. Interesting studies of dynamic stall have also been conducted by UTRC [58] with hot films.

Local velocity measurements can also be made in the vicinity of the blades and in the rotor wake. In hover, they can be used to evaluate the local blade circulation thereby replacing the need for pressure measurements which are virtually impossible to make on small scale models. They also provide an accurate description of the induced velocity field and of the tip vortex trajectories. Laser velocimeters are

Fuselage static Instability

A bare fuselage is intrinsically aerodynamically unstable, and tail surfaces are added to act as stabilizers. On a helicopter, however, these surfaces generally operate in the disturbed flow shed by the rotor hub and the fuselage afterbody. The efficiency of the tail fin and horizontal stabilizer is thus severely reduced because of the dynamic pressure losses inside the wake [166,118]. The same is true of the tail rotor contribution to the directional (wheathercock) stability. Figure 75 from [166], shows dynamic pressure distributions measured in the vicinity of the tail rotor on a BK-117 in climb, descent and level flight. The changes which can be observed in the wake structure and position relative to the stabilizers are due to differences in the aircraft angle of attack and in the rotor induced velocities. In descent the tail fin is entirely immersed within a region of low dynamic pressure which probably corresponds to the heart of the rotor head wake.

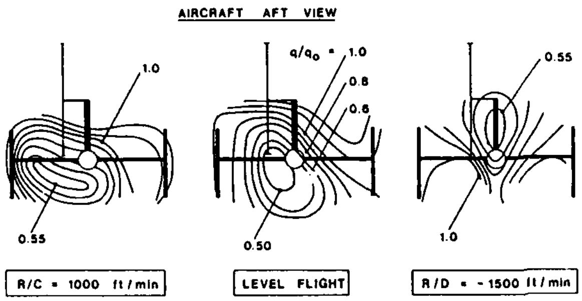

Fig. 75. Wake displacement with flight conditions (MBB/KHI BK-117, flight tests), from [166]

Inadequate airframe stability may be improved in two ways: by increasing the efficiency of the stabilizer surfaces or by acting on the dynamic pressure losses in the wake.
In the first hypothesis, the size of the stabilizer may be increased, with corresponding weight and drag penalties, or special stability augmentation devices may be implemented such as those described in [123].
If one chooses to act on the dynamic pressure losses in the vicinity of the stabilizers, this can be done either by reducing the strength of the wake shed by the rotor head and the fuselage afterbody or by deviating the wake away from the stabilizers. The first solution is certainly preferable, since any reduction in the wake intensity induces a corresponding drag reduction [123]. Unfortunately, this is not always feasible, and it is then necessary to redirect the wake, at the risk of simply changing the flight configuration for which the problem occurs. Indeed, as shown in figure 75, the position of the wake relative to the tail surfaces depends on the flight configuration so that the wake displacement might be beneficial in one flight configuration and detrimental in another.

The effectiveness of the proposed stabilizer modifications can be checked by comparing the longitudinal and lateral aerodynamic characteristics of the modified and original aircraft [123]. Wake modifications (attenuation or deviation) are best evaluated by probing the flow in the neighborhood of the tail surfaces. Any means of reducing the rotor head drag are effective in attenuating the wake, for example the pylon fairing of [123] which simultaneously reduces the dynamic pressure losses and the drag. A commonly implemented technique to deflect the wake downwards below the stabilizers is to install a convex fairing on top of the rotor hub [108,166,123].

Tail Shake Dynamic Instability

This problem is frequently encountered during aircraft development. It is seldom detected during wind tunnel tests, and is generally revealed in flight [108,166,109].

The phenomenon involves both helicopter aerodynamics, from which the excitation originates, and the airframe structural characteristics which determine the response modes whose natural frequencies are generally close to one of the lower harmonics of the rotor rotation speed Ω. This duality makes it difficult to analyze the problem in the wind tunnel because the dynamic similarity of the model is not always assured. The aircraft development work may be done in flight [166], but wind tunnel tests are also conducted in an attempt to determine (and eliminate) the causes of the tail shake by analyzing the airflow [108,109].
The problem here is no longer static but dynamic, and concerns the unsteady flow components. The studies in this area are all based on the spectral analysis of the 3-D velocity fluctuations in the wake the objective being to identify the sources of excitation in the flowstream capable of inducing low-frequency vibrations of airframe components.
The wake turbulence is distributed over the entire frequency spectrum with higher energy levels at low frequencies [109]. To this turbulent spectrum the rotation of the rotor head adds discrete energy peaks at harmonics of the rotation speed, especially at integer multiples of the hub shank passage frequency bΩ [108,166,109]. Energy concentrations have also been recorded at non-harmonic frequencies [166,109] appearing at constant Strouhal numbers which could be related to vortex shedding by large unstreamlined airframe components such as the spoiler in [124]. This spoiler, which had been installed at the rear of the fuselage of the BK-117 to improve directional stability, generated large vortices which appeared in the power spectral density of the fuselage pressure signals near the tip of the spoiler as a concentration of energy centered around 31 Hz. The removal of the spoiler greatly improved the tail shake situation [124].
In [108], efforts to define configurations with reduced low frequency wake disturbances also led to a satisfactory solution of the tail shake problem encountered on the BOEING-VERTOL YUH-61A UTTAS.
Finally in [109], a pylon fairing that proved to be effective as regards tail-shake on the AS-365N not only reduced the rotor head drag and the low frequency turbulence of the shed wake but also attenuated the discrete energy peaks at rotor harmonic frequencies when it was tested on the AS-355 TWINSTAR. In this analysis the tail fin root bending moments were used rather than hot film signals acknowledging the fact that the tail fin integrates the unsteady airloads produced by the fluctuating surface pressures and that the power spectral density of the fin bending moment exhibits phenomena similar to those observed on hot film signals placed in the airflow in the vicinity of the fin.

Each of the experimental methods implemented thus led to results that were confirmed in flight. However the underlining mechanisms of tail shake instabilities are not yet fully understood. To what extent do they depend on the low frequency turbulence in the wake, on the discrete velocity fluctuations at rotor harmonic frequencies or on other vortex shedding phenomena occuring at fixed Strouhal numbers? Many further tests will be required before these questions are answered.

2.1.4 CONCLUSION ON EXPERIMENTAL FUSELAGE AERODYNAMICS

Fuselage aerodynamics today requires the use of experimental methods, and primarily wind tunnel tests. A number of precautions are essential to obtain valid results: (1), a thorough analysis of the interference due to the model support structure, (2), careful attention to the accuracy of the model and to the Reynolds number, especially for the rotor head, and (3), allowance for airstream blockage effects when the model cross section represents an appreciable fraction of the tunnel section.
With due consideration for these points wind tunnel tests of small scale models provide results that correlate well with full scale tests and which can be used in flight.

Two critical areas are of particular importance with regard to aircraft drag:

- the fuselage afterbody section which can double the fuselage drag if poorly designed; two types of flow may occur in this area, and the transition from one to the other is accompanied by a sharp change of drag

- the rotor head, whose drag is highly dependent on the shape of the cowlings beneath it.

The afterbody section drag may be reduced by adopting favourable geometric parameters (contraction ratio and camber), or by improving on unfavourable parameters through the use of special devices (spoilers, deflectors, etc.). Concerning the rotor head significant drag reductions can be achieved with compact hubs and properly designed hub and pylon fairings.

Fuselage and rotor head aerodynamic interactions with the tail unit have a strong influence on the static stability of the airframe and may lead to tail shake dynamic instabilities which can be investigated with experimental methods. Drag reductions attenuate the wake intensity and generally alleviate the stability problems related to these interactions. However, by permitting higher airspeeds, they may in turn lead to further stability problems.

2.2 THEORETICAL METHODS

A second approach in the field of aerodynamic research became available in the 1960's with the introduction of a new and more powerful generation of mainframe computers (IBM 360, UNIVAC 1100, CDC 3600) that made it possible to implement theoretical methods for which the aerodynamic and numerical basis had only recently been established [125,126]. Computational methods in fuselage aerodynamics were thus developed, first by fixed-wing aircraft manufacturers, and only became available for helicopter research in the early 1970's. Helicopter manufacturers were thus offered proven computational codes on a second generation of more powerful computers based on a new technology. The aircraft configurations involved were however more complex as helicopter fuselages tend to be considerably less streamlined than fixed-wing aircraft.

The principal calculation methods implemented from this period to the present time are reviewed here with an assessment of their aerodynamic justification and their limitations. This review is followed by a prospective study of the potential developments that can be expected of these methods.

2.2.1 The Theoretical Problem

For most helicopters, the airflow around the fuselage (considered alone) can be separated into three distinct zones (figure 76):

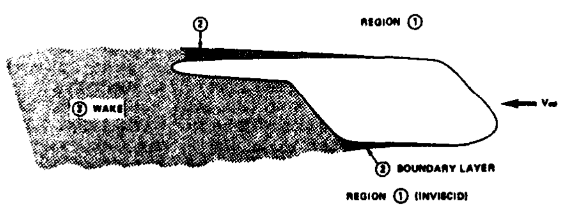

Fig. 76. Schematic of flow regions over a helicopter fuselage

- Zone (1), in which the fluid is considered to be ideal and irrotational. Viscosity and turbulence effects are negligible, and the airflow may be described by the EULER and BERNOULLI equations:

  $\mathrm{div}(\phi) = 0$

  $V = \mathrm{grad}(\phi)$

  $p + \rho V^2/2 =$ constant

  At the airspeeds commonly flown by helicopters today, the air may be considered, for practical purposes, as an incompressible fluid (ρ= constant).

- Zone (2), the boundary layer, a thin but highly important region of the flow where viscous friction on the fuselage surface originates and in which turbulence grows as the flow moves downstream. The local equations describing the behavior of the boundary layer thus include exact terms representing both viscous and turbulent friction.

- Zone (3), a turbulent wake arising from boundary layer separation. Viscosity has little influence on the airflow here, which is dominated by the effects of turbulence and principally by vorticity giving rise to rotational motion.

On a real helicopter, the airflow around the fuselage involves other complex airflows related to the presence of the engines and rotor:

- Engine air intake suction flow

- Engine exhaust flow characterized by turbulent and vortical flow

- Rotor flow, characterized by a vortex sheet relatively well organized in comparison with the wake shed by the superstructure and fuselage afterbody.

From a formal standpoint, the evolution of these flows (fuselage, rotor, engines) is governed by the time dependent NAVIER-STOKES equations or, if only a pure stationary description is required, the averaged NAVIER-STOKES equations which include turbulent stresses for which closing relations must be developed.
Solving these equations is a monumental task that is today only conceivable in the case of very simple geometries, but in no way for a complete helicopter. These methods are heavy and very costly in terms of computer time, and are as such ill suited for routine industrial use.
Faced with the complexity of the general problem, the engineer must then fall back on approximate methods based on a simplified description of the aerodynamic field. Turbulence is most often neglected outside the boundary layer, and free flow streams are considered as irrotational or rotational ideal fluid flows.
Under these conditions, what can be expected from a theoretical approach? Mathematical simulation is not necessarily a direct competitor of wind tunnel testing, and for relatively simple methods two immediate applications can be considered :

- Analysis and better understanding of experimental airflows

- Examination of a large number of geometrical shapes and selection of the most promising ones for wind tunnel testing.

In addition to this experimental supporting role, there is a genuine need in the industry for more sophisticated methods capable of providing valid predictions of fuselage aerodynamic coefficients, and especially of the drag coefficient. The ideal for an aircraft manufacturer would be to dispose of a fully computerized system covering everything from computer-assisted design methods for generating fuselage shapes to computational methods for estimating aerodynamic loads.
The following section examines the extent to which today's computational methods meet these requirements.

2.2.2 Evaluation of Existing Computer Methods

Most of the computer codes developed in the last 20 years for fuselage aerodynamics are based on the method of singularities. Although the use of singularities is not recent in aerodynamics (cf. PRANDTL's use of vortices in his lifting line theory) it was not until about 1960 that a numerical approach was introduced in the work of SMITH & PIERCE [125] and of HESS & SMITH [126].

The basic idea is to simulate an irrotational ideal fluid flow around a body by placing a set of singularities outside the fluid region, i.e. inside the body. Each singularity creates in the fluid region a potential verifying EULER's equations. The most delicate, in reproducing a given flow field, is to select the singularities so that NEUMANN's condition (fluid slippage along the wall) is met at every point on the surface of the body. This well-known theory will not be examined in detail here. Nevertheless, it is worth considering a number of basic points.
A classical result of irrotational incompressible fluid flows is d'ALEMBERT's paradox, according to which the net force acting on a body of arbitrary shape is zero, provided the flow is uniform.
Under these conditions it is clear that a body modelled with "source" and "sink" type singularities can be subject neither to lift nor to drag. In order to obtain a non-zero resultant of aerodynamic forces, other singularities must be introduced in the flow, such as vortex rings, which violate the spatial uniformity of the potential and thus get around d'ALEMBERT's paradox.

A numerical implementation of the singularities method requires that a fuselage (and/or a wing) be broken down into a number of flat panels as shown in figure 77.

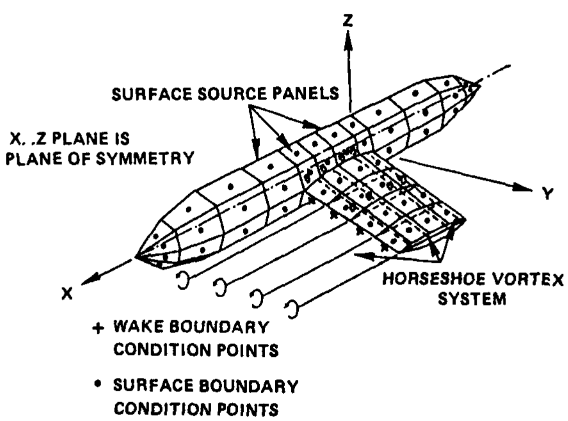

Fig. 77. Schematic of source and vortex panel potential flow model, from [134]

In the case of a fuselage, the simplest approach is to assign an initially unknown constant density of sources to each panel, and to postulate NEUMANN's condition at each panel control point. The solution of the problem, i.e. the determination of the unknown densities, is thus reduced to solving a linear system of equations.

Wing modelling generally involves a distribution of horseshoe vortices representing the circulation of the flow around the airfoils and the vortex sheet shed by the trailing edge. It may be noted that the actual airflow is rotational inside a thin layer downstream from the trailing edge. Moreover, the method of singularities only determines the velocity potential (and therefore the velocity) in regions of irrotational flow. In order to apply such a method to lifting configurations it must therefore be assumed that the vortex phenomena are concentrated in a vortex sheet of zero thickness.

Compared with other numerical methods in aerodynamics using finite differences or finite volumes, the method of singularities presents the enormous advantage of offering a description of the velocity field at any point in space (except at the singularity locations) at the cost of a relatively simple discretisation limited to the surface of the body being studied. This feature explains the widespread industrial use of the method.

2.2.2.1 First Applications of Potential Calculations to Helicopter Fuselages

The first attempts at applying potential calculations to a helicopter fuselage were largely indebted to the DOUGLAS-NEUMANN Program, discussed in [126], which uses source type singularities. Examples include the work of MONTANA [127] in 1973, SHEEHY [128] in 1975, GILLESPIE [129] in 1973 and GILLESPIE & WINDSOR [130] in 1974.
No attempt was made at that time to model separation zones at the rear of the fuselage, and the validity of the results was thus seriously limited. Some authors [127], aware of this shortcoming, reserved this type of analysis to highly streamlined configurations unlikely to produce boundary layer separation. It soon became apparent that the knowledge of the pressure field alone on the fuselage surface was not sufficient to understand the flow behavior, hence the interest in supplementing the potential calculation by a fuselage streamline calculation module.
One particularly interesting approach [129] may be broken down as follows:

- Calculation of the potential flow around the fuselage using the DOUGLAS-NEUMANN program
- Calculation of the streamlines on the fuselage surface using a code taken from [131]
- Calculation of the boundary layer on each streamline assuming small transversal flow. The actual streamline is thus replaced by a fictive line with the same pressure distribution which is assumed to be developing in axisymmetric flow on an axisymmetrical body. The divergence of the actual streamlines is equivalent to a variation in the radius of the fictive body. In any event, this is a 2-D boundary layer calculation.

This method was applied to the fuselage of the BO-105 (figure 78) for which detailed experimental results were available. The correlation between calculated and test results is shown characteristically on figure 79, which represents the pressure coefficient along the lower aircraft centerline in the symmetry plane ($\alpha=-15°$). Extremely close agreement was obtained for the nose and the central fuselage portion, but the theoretical prediction becomes inaccurate in the afterbody area: the potential calculation predicts a strong recompression which in fact does not occur as a result of boundary layer separation. The preceeding example highlights one of the major obstacles to any attempted theoretical prediction: the fuselage afterbody often produces regions of flow separation where turbulent eddy flows are present which no longer follow the laws of ideal irrotational fluids. This is precisely the reason why real flows generate pressure drag which cannot be predicted by a simple potential calculation using source-type singularities.

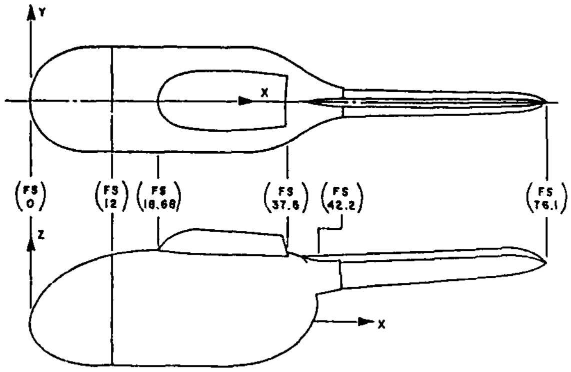

Fig. 78. Sketch of BO-105 fuselage, from [129]

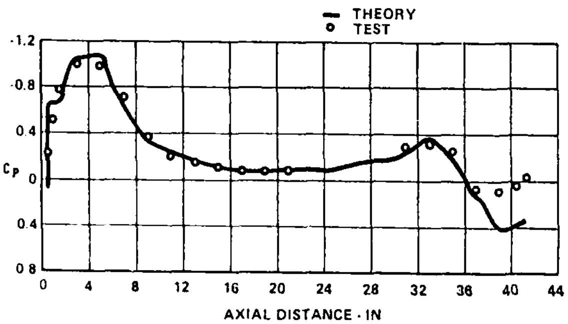

Fig. 79. Pressure coefficient vs. axial distance on bottom centerline of BO-105 model at α=-15°, from [129]

Despite these drawbacks, a number of authors have successfully used numerical codes on the same principle as the one described in [129], including CLARK & WILSON [132] and VENEGONI, MAGNI & BALDASSARINI [133].

In [132] a source-type singularities method associated with a streamline calculation module and a 2-D boundary layer method developed by AMI is used to determine the flow separation line in the afterbody area of different fuselage shapes. Although drag calculations are impossible, the theoretical analysis provides a qualitative shape optimization criterion, since the drag coefficient is closely related to the size of the separated region.

Reference [133] discusses the use of a similar computer program in the design of the A-109 fuselage. Although no spectacular discoveries were made, the analysis produced a better understanding of the experimental results and allowed the effects of shape modifications to be quickly evaluated by plotting the fuselage streamlines (figure 80).

Fig. 80. Streamline distribution over the A-109 model, from [133]

2.2.2.2 Development of Separated Flow Models

By the mid 1970's, despite the encouraging results mentioned above, it became clear that a productive use of theoretical methods at the preliminary design stage required a better assessment of the airflow downstream from the separation lines. Without a suitable wake model, the theoretical methods would undoubtedly remain a useful but secondary design tool compared with wind tunnel test results.

In order to fully appreciate the difficulty of the problem, a closer look is required at the phenomenon of boundary layer separation. For greater clarity, consider the separation of an incompressible 2-D boundary layer subjected to an adverse pressure gradient (figure 81). At the high Reynolds numbers characteristic of helicopter flight, the boundary layer is naturally turbulent and obeys to the following equations:

- Continuity equation:

$$\frac{\partial u}{\partial x} + \frac{\partial v}{\partial y} = 0$$

- Momentum equation:

$$u\frac{\partial u}{\partial x} + v\frac{\partial u}{\partial y} = -(1/\rho)\frac{\partial p}{\partial x} + \frac{\partial}{\partial y}\left(\nu\frac{\partial u}{\partial y} - \overline{u'v'}\right)$$

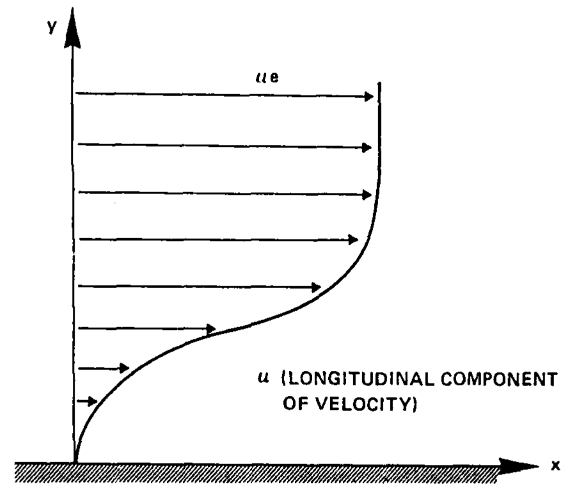

Fig. 81. Typical sketch of the longitudinal velocity profile in a 2-D boundary layer near separation

Where:

ν: kinematic viscosity

p: static pressure (assumed constant throughout the boundary layer thickness)

ρ: density

u', v': turbulent velocity fluctuation components

u: longitudinal velocity component

v: normal component

In the momentum equation in addition to the viscous friction term $\nu\partial u/\partial y$ which is significant only in the immediate vicinity of the wall surface, there appears a turbulent friction term materialized by the turbulent stress factor $-\overline{u'v'}$. This term, amplified by the adverse pressure gradient, contributes to the deformation of the mean velocity profile. For example, in the logarithmic region of the velocity profile, the turbulent velocity fluctuations commonly reach 10 % of the external velocity Ue.

At the nose and on the main fuselage section, the boundary layer thickness is negligible compared with the body dimensions thereby justifying the use of an ideal fluid singularity method. This is no longer true near the separation zone, however, where both the turbulent kinetic energy and the vorticity (represented by the $\partial u/\partial y$ term) are distributed in a 3-D volume and are no longer confined in a thin layer at the surface of the fuselage. Downstream from the separation zone the turbulence released into the wake evolves under the effect of the mean velocity gradient and in return reacts on the mean velocity field. Its intensity gradually decreases as energy is transferred from large-scale turbulent structures to small-scale structures which dissipate through viscosity.

An interesting description of a fuselage wake is given by the vortex stretching theory [157]. At high Reynolds numbers it is assumed that the behavior of most vortex sheets (i.e. instantaneous vortex line envelopes) is governed by the time-dependent ideal fluid equations, while viscosity is involved only in very small scale motions. According to KELVIN's circulation theory, the vortex lines (and therefore the vortex sheets) are conserved during the evolution of the fluid in time, although they may be distorted. Thus, a fluid particle may be subject to totally random motion while moving along the vortex sheet. Moreover, a particle subjected to random motion tends on average to move away from its point of departure. Similarly, the distance between two particles tends on average to increase with time. If these particles are situated at the ends of a vortex line segment, the length of this segment tends to increase. A vortex tube thus has a tendency to stretch out axially, while becoming thinner to ensure mass conversation. Conservation of the kinetic moment ωr of a vortex tube implies that the rotational speed ω increases as the radius r decreases. The process continues until the tube of very small radius is finally dissipated by viscosity. The overall result is the gradual diffusion and disorganization of the wake as it moves downstream away from the fuselage.

As can be seen the general evolution of the curl of the velocity poses a complex problem. If the effects of viscosity and turbulence in the wake are neglected, then this evolution is governed by HELMHOLTZ' vorticity equation:

$D\omega/Dt = \omega.\mathrm{grad}\ V$

Where D/Dt is the particle derivative, V the velocity, and $\omega = \mathrm{rot}\ V$.

Consider now the solutions proposed to date for modelling the wake behind helicopter fuselages. Two broad categories may be distinguished. One is a set of models developed by AMI and based on the following assumptions:

- Wake turbulence is not taken into account
- The vorticity is concentrated in a region of zero thickness forming a stationary vortex sheet.
- The vortex strength remains constant along the sheet and is determined by its value on the separation line.

Except for the very first model in which the vortex sheet was cylindrical [134] (figure 82) the program includes several iterations designed to align the sheet on the local flow direction. The vortex sheet, which in the final stage is no longer crossed by the fluid, becomes a slippage surface, one of the two discontinuities compatible with ideal fluid equations (the second type being the shock wave).

Reference [139] discusses the application of a model reserved for non-lifting configurations in which the body is represented by a distribution of sources with constant panel densities. A characteristic example illustrating the possibilities of this method is given by the calculation of the airflow around a sphere (figure 83). Compared with the potential flow calculation without wake modelling, the vortex sheet modelling brings a qualitative improvement in the heart of the separated zone.

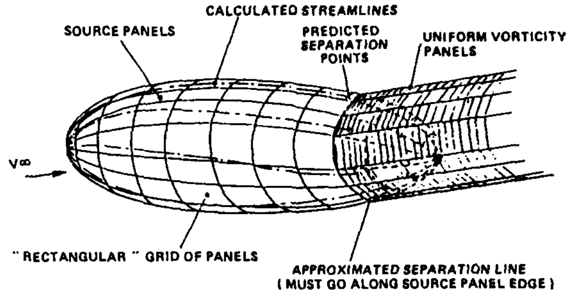

Fig. 82. Schematic of interim model for modeling separated flow on bodies, from [134]

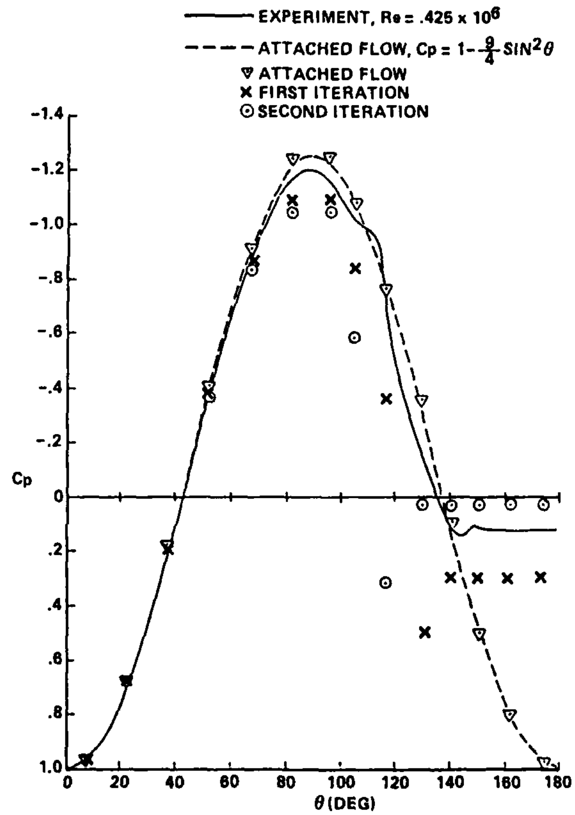

Fig. 83. Comparison of measured and computed pressure distributions on a sphere, from [134]

Nevertheless, reliable drag predictions are still not possible, as shown by the discrepancy between the computed and experimental pressure coefficients. With the new model, the agreement even breaks down completely near the junction between the sphere and the vortex sheet.

The authors then concentrated on lifting configurations (wings of finite span at high incidence) in which the body is represented by distributions of sources and doublets. The problems of interaction between the body and the vortex sheet are then reported to be less critical. Advanced techniques have even been developed to increase the mesh density on the body in the areas of strong interaction [140]. To our knowledge, however, no satisfactory correlation with experimental results on rounded bodies such as the sphere has yet been presented.

Indeed, few results concerning isolated fuselage wake modelling have been published, whereas the literature includes many studies analyzing the influence of the fuselage on external components such as the rotor head [136,137,138].

It should be noted here that the validity of the representation of a wake by a single vortex sheet has never been convincingly established. This type of model is perfectly suitable for the wake shed from the trailing edge of a wing of finite span which physically resembles a sheet, but may be inadequate in the case of a thick boundary layer separating from the rear of a rounded body. The poor results shown on figure 83 in the flow separation area could then be attributed to the underlying assumptions of the model itself rather than to the nature of the singularities used.

In order to avoid the inconvenients of representing a volume distribution of vorticity by a sheet of zero thickness, MBB has developed a second category of models based on a volume distribution of vortex singularities [141,142]. Derived from a potential flow code using source-type singularities [143], the most recent version [142] achieves a 2-D boundary layer calculation along several streamlines at the surface of the fuselage and determines a flow separation line. The originality of the method lies in the construction of a 3-D wake mesh downstream from this line, in which vortex lines are distributed (figure 84). The radial and longitudinal evolutions of these vortices are determined empirically from experimental results obtained with rounded bodies such as the sphere.

The advantage of this type of model is to offer a more refined representation of the rotational component of the mean flow than can be obtained with a single vortex sheet but at the same time it also raises a number of problems:

- The vortex distributions, based on empirical laws, cannot be generalized and become increasingly unsuitable as the flow deviates from the original axisymmetrical flow conditions

- The method does not take into account the presence of the aircraft tail boom, and no provision is made for the resulting wake deformation: the fuselage must therefore be truncated at the root of the tail boom for calculation purposes (figure 84)

- No clear theoretical background has been advanced for the volume distribution of singularities. In the theory of singularities it is required that the mathematical (and physical) singularities be located outside the fluid region, i.e. inside the body or, if otherwise unavoidable, in areas of zero thickness within the fluid region that constitute discontinuity or slippage surfaces. The laws of ideal fluids, which postulate the existence of such surfaces, are thus observed throughout the fluid region. In the case of the model considered here, it is difficult to assimilate the tangle of vortex line segments with a representation of slippage surfaces.

Figure 85 compares the model predictions with the experimental results for the airflow around a sphere. The results are similar in certain respects to those given by the A.M.I vortex sheet model i.e. (1), the pressure level is correct in the wake, and (2), the recompression is overestimated near the flow separation line at the boundary of the wake. In final, the model leeds to a sizeable error on the pressure coefficient and cannot be expected to provide valid drag predictions even for the relatively simple configuration that was used as the basis for its development (prescribing the radial and longitudinal vortex distributions).

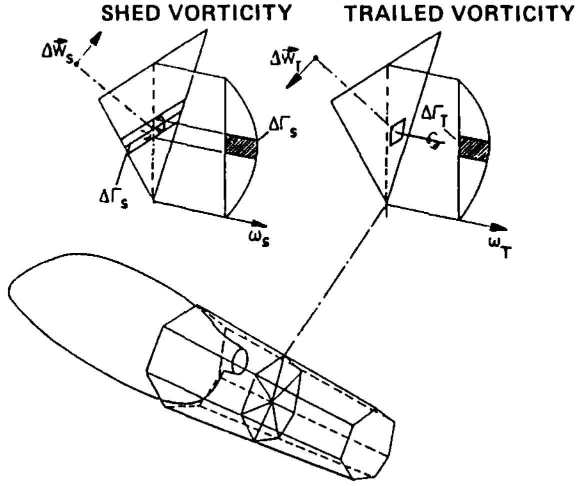

Fig. 84. Calculation of wake induced velocities, from [142]

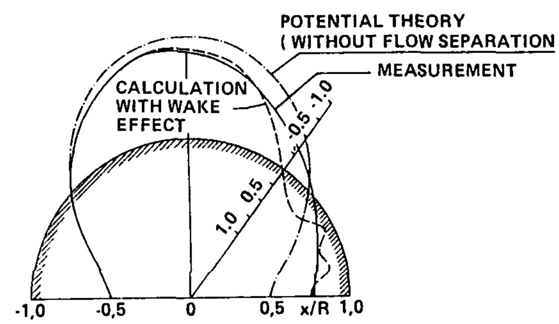

Fig. 85. Measured and computed pressure distributions on a sphere ($Re=0.45x10^6$), from [142]

2.2.2.3 Complex Configurations

The difficulty of making valid drag predictions being recognized, the trend since 1980 has been to move away from attempts to obtain accurate drag estimations on simple configurations (isolated fuselages) and instead to concentrate on more qualitative studies of complex configurations. These may include models of the rotor head, rotor disk, powerplant airflows, tail rotor and even of the tail fin and stabilizer assembly, but the most frequent combination is the fuselage and rotor.
Both aspects of the mutual interaction may be considered:

Rotor downwash on the fuselage

Since the aerodynamic characteristics of the fuselage alone cannot be accurately predicted, these results can only be very approximate. Moreover, fuselage aerodynamics are of concern primarily at high speeds, when the rotor induced velocities become small compared with the helicopter airspeed. This aspect is therefore of limited interest. Reference can be made nevertheless to the work of FREEMAN [144] in which the fuselage is modelled by sources and the rotor wake by a set of vortex lines. What limits his approach is the fact that he uses a prescribed wake semi-empirical rotor model which modifies the airflow over the fuselage but is not in return affected by it.

Fuselage effects on rotor operation

Efforts to improve streamlining by minimizing the fuselage cross section have often led helicopter manufacturers to design extremely flat transmission systems in which the rotor is dangerously close to the fuselage (e.g. WG-13 LYNX, UH-60 UTTAS). The passage of a rotor blade above the fuselage can produce a sudden variation of the blade angle of attack and induce undesirable structural vibrations. This situation was studied in detail by CLARK & MASKEW [145,168] with a vortex sheet model developed by AMI (VSAERO code). In view of what has already been said about the inadequacies of modelling an arbitrary wake by a vortex sheet, a reliable description of the airflow around a complete helicopter is not to be expected. Nevertheless, this approach sets a new record for complexity in implementing a method of singularities (figure 86). Successive iterations take into account: (1) the influence of the rotor flow field on the airflow around the fuselage, (2), the influence of the fuselage on the development of the vortex sheets shed by the rotor, and (3), if required, the influence of additional components such as rotor head, tail rotor, horizontal and vertical stabilizers.
The rotor is represented by a vortical disk with vortices trailing at the blade tips. The fuselage is modelled by source and doublet distributions, and the vortex-wall interactions are processed as described in [140]. This program has revealed the importance of the angle of attack variations on the blades as they pass over the fuselage, and the adverse effect of the rotor head on the airflow around the aft blade.

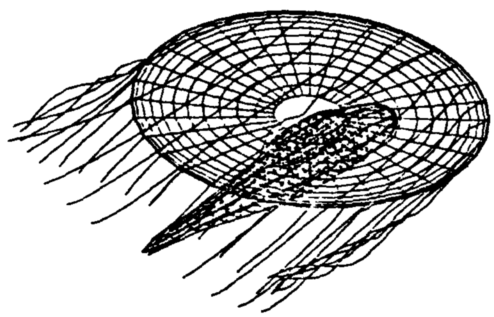

Fig. 86. Basic panel model for the study of rotor/body interactions, from [145]

Among the authors which have delt with the interference created by the fuselage on the rotor we may also mention WILBY, YOUNG & GRANT [165], HUBER & POLZ [166], JOHNSON & YAMAUCHI [167].

An important problem when dealing with complex configurations is the huge number of panels which are necessary. With the present limitations of computer memory size it is not possible to discretize simultaneously the rotor and fuselage with a sufficiently fine mesh to reveal, for example, the higher-order harmonics required to analyze dynamic rotor loads.

As a conclusion to this review of existing methods, it should be recognized that the most recent developments, i.e. programs capable of dealing with multi-component configurations, resemble very much demonstration exercises of the possibilities of the theoretical approach, which require each time partial rewriting of existing programs. The routine use by project engineers of a reliable code to evaluate the consequences of specific airframe modifications is still a remote prospect.

With the progress made in recent years concerning the generation of calculation meshes, Computer Assisted Design methods now make it possible to modify a geometry and arrange the corresponding panels in about half a day. In order to become a serious competitor for wind tunnel testing, numerical methods must first be capable, in a similar time interval, of furnishing a valid quantitative estimate of the shape modification, even if initially only relatively simple shapes are considered. An effort is thus required to provide the computational methods with a more rigorous theoretical basis, and ultimately with the experimental validation that they are still lacking.

2.2.3 Towards a more Fundamental Approach

If the objective of the research in computational aerodynamics remains the valid prediction of fuselage characteristics (e.g. aerodynamic coefficients), then the problem must be reconsidered from the outset by concentrating the research effort on the isolated fuselage. A critical analysis is required of the methods discussed above in order to identify any overly restrictive or even false hypotheses.

Three modelling levels shared by all of the previous methods can be a priori reconsidered: (1), the potential flow calculation in the irrotational flow region upstream from the separated flow areas, (2), the 2-D boundary layer calculation along the fuselage streamlines, and (3), the wake calculation downstream from the separation zones using a singularities method.
In the following sections we propose to discuss the limitations of each of these models.

2.2.3.1 Limitations of Potential Calculations in Irrotational Flow

The problem is an exact one (EULER's equations) and does not involve modelling per se (in the usual sense of the word). The difficulty lies in its numerical resolution, for which a number of choices must be made: (1), the type of singularity (source, doublet, vortex), (2), the type of distribution (uniform or higher-order panel distributions), and (3), the size of the mesh.
Practical experience has shown that very good agreement with experimental results can be achieved for the pressure on the fuselage in the unseparated zones by means of a simple distribution of sources with uniform panel density. The number of panels permitted by existing programs (about 1000 panels per half-fuselage) is enough to ensure this and allows the aircraft geometry to be correctly specified. For a given number of panels, the quality of the mesh is considerably enhanced by the use of the block mesh technique developed by AMI (with variable mesh density between blocks). Moreover, it has been verified that, for a simple potential calculation on a non-lifting body, the use of doublets gives results extremely close to those obtained with source-type singularities.

As a general rule, the potential calculation in the regions of unseparated flow is the most reliable step in all of the previously considered methods.

2.2.3.2 Limitations of Streamline Boundary Layer Calculations

The first computer codes dealing with fuselage aerodynamics were developed at a time when the only boundary layer methods available in industry were 2-D. A few years earlier, in 1968, the STANFORD conference on boundary layer calculations [164] had established the validity of several 2-D integral methods, and it was only natural to attempt to adapt these to potential flow codes. Their application, based on the assumption of minimal transverse flow, made it necessary to compute streamlines on the fuselage surface.
Subsequently, in the late 1970's, 3-D integral boundary layer methods became available and were in turn validated by various workshops on 3-D boundary layer calculations, including STOCKHOLM [155] in 1978 and AMSTERDAM [156] in 1979. At that time however, the fuselage aerodynamic codes were already relatively well structured and were not called into question. Present-day use of 2-D rather than 3-D boundary layer methods is therefore more likely to be a consequence of the chronology of program development rather than the result of a critical analysis of the aerodynamic problem.

Useful information on the nature of 3-D boundary layers and the relevant computational methods can be found in references [146] through [151]. Only a few fundamental aspects will be recalled here.

- The 3-D character of the boundary layer is highly dependent on the curvature of the streamlines on the fuselage. In the case of the flow around an axisymmetrical body at zero incidence, the purely longitudinal curvature of the streamlines does not induce any 3-D effects. Conversely, when the same streamlined body is placed at a certain incidence in the flowstream, a transverse curvature appears, generating strong 3-D effects in the boundary layer.

- A 3-D boundary layer is characterized by a change in the direction of the speed vector across the boundary layer between the exterior and the surface of the body. The limit of the direction of the speed vector on the surface of the body determines a skin-friction line (as opposed to the inviscid streamline).

- The principal difference between 2-D and 3-D boundary layers concerns the separation phenomenon. 2-D separation is usually defined as the disappearance of viscous friction and the reversal of the surface flow direction. In the 3-D case, the notion of separation is not clearly defined. Nevertheless, experimental visualizations and theoretical methods [149] often indicate a concentration or focalization of the skin-friction lines along a curve, the envelope, which may be assimilated with a separation line, and which is in fact a line of singularities for the theoretical methods.

The theoretical predictions of the separation line vary widely according to which type of method is used. Figure 87 shows the skin-friction lines on a streamlined body at 10° incidence, as calculated by AEROSPATIALE with the integral 3-D boundary layer method developed by COUSTEIX & AUPOIX [150] (the inviscid velocity field was determined using a source-type singularities method). Very distinct 3-D separation can be observed in the focalization of the surface flow lines. Under the same conditions, a conventional 2-D boundary layer calculation along surface streamlines would only predict separation on the upper line of the body in the plane of symmetry.

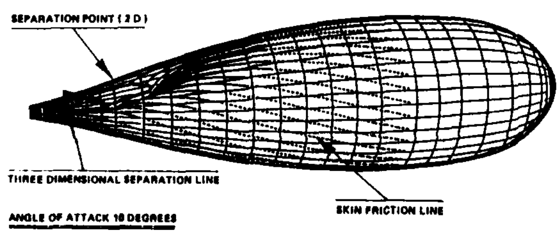

Fig. 87. Comparison of 2-D vs. 3-D separation lines on a streamlined body at $\alpha=10°$

On a helicopter fuselage, the localization of possible separation is of considerable importance since it determines the size of the low pressure region behind the fuselage which is the principal source of drag. A crucial question then arises: are the flowstreams routinely encountered around helicopters likely to induce separation phenomena with strong 3-D effects thereby requiring the use of 3-D boundary layer methods ?

A study undertaken at AEROSPATIALE based on the experimental results published in [152] shows that in the case of a fuselage with a strong afterbody contraction, the separation lines obtained with 2-D and 3-D methods are different but qualitatively comparable, the 3-D calculation providing slightly better agreement with experimental results. It is entirely another matter, however, when dealing with streamlined bodies at non-zero incidence such as depicted in figure 87.

Broadly speaking, a 2-D method is inherently unreliable in that the user cannot readily detect the point beyond which the results become false. The performance of the fuselage codes described above should thus be significantly improved by incorporating a 3-D boundary layer method. Moreover this integration does not pose an insurmountable task as demonstrated by the recent work of HIRSCHEL at MBB [153,154].

Anticipating on the next paragraph, it must be emphasized that the determination of a valid separation line is an absolute prerequisite for wake calculations, irrespective of the wake model used, since this line in turn determines the initial conditions from which the wake will develop. From an operational standpoint, it is easier to define the separation line from a 3-D calculation with a complete fuselage mesh than with a 2-D method in which the fuselage must be covered by trial and error with numerous streamlines whose final destination is unknown in advance.

2.2.3.3 Wake Calculations

It was shown in the previous section that the principal limitation of the wake calculation (particularly near the separation area) came from the use of a surface model to represent a 3-D volume distribution of vorticity. More generally while the true unsteady behaviour of vortex sheets can indeed be described in terms of an ideal fluid, the time-averaged velocity field (assuming that a statistical average exists) corresponds to a volume distribution of vorticity that can hardly be represented by a single stationary vortex sheet. Nevertheless, two basic reasons have led authors to use a method of singularities often with complete disregard for the physical reality of the phenomenon: (1), the desire to terminate the calculation of the aft fuselage section with a method already present in the programs, and (2), a reluctance to use numerically heavier methods on an industrial basis.

Setting aside simulations of the time-dependent NAVIER-STOKES equations which are still limited to low turbulent Reynolds numbers, the most efficient means now available for computing complex flow fields is a finite difference solution of the averaged turbulent NAVIER-STOKES equations. Such purely elliptical calculations are capable of predicting phenomena as complex as the horseshoe vortex located in front of a compressor blade root, or in front of the wing-body junction on an aircraft [158].

The introduction of turbulence into the monumentum equation, generally in the form of a turbulent viscosity term, remains a perfectible aspect of the method.

The high computer time required for an elliptical resolution of the NAVIER-STOKES equations has held some authors to prefer faster semi-elliptical methods [159,160,161] capable of dealing with a descending step or a fuselage afterbody, and also, but to a lesser degree, with downstream obstacles (e.g. the airflow over an ascending step). In any event, the preparation of a 3-D mesh of the fluid region, the accurate determination of the boundary conditions, and the computer time required are not yet compatible with routine industrial applications.

An intermediate solution is required to fill the gap between the "fast" methods providing only qualitative results and the more productive but time-consuming codes.

One possibility that will be considered at some length here is currently being evaluated by AEROSPATIALE: an ideal fluid model of vortical wakes by means of point vortices. This type of model was studied in detail by REHBACH of ONERA [162,163]. As mentioned earlier, the evolution of the curl of an ideal fluid is governed by HELMHOLTZ'equation:

$$D\omega/Dt = \omega.\text{grad } V$$

Unlike conventional vortex sheet methods in which the vortex strength is held constant and equal to the value it had when it was shed by the boundary layer *during separation, the time-dependent evolution of* the vorticity can be computed here by means of an exact equation. It naturally applies to unsteady aerodynamic flows, but can also be used to compute stationary flows by progressively converging towards a steady-state solution. The slipstream is no longer considered as a sheet, but rather as a set of point vortices. At each time step, point vortices are emitted along a separation line (or, as in the figure below, *along the leading edge of a wing at high angle* of attack) where they represent in concentrated form all of the surrounding vorticity. The vortices then propagate by convection according to HELMHOLTZ' equation, in the same way as all the previously emitted vortices (figure 88). It should be noted that a point vortex considered alone has no physical meaning. Only complete vortex rings are acceptable in an ideal fluid and create a true velocity potential.

However, the set of point vortices effectively represents a vortex sheet.

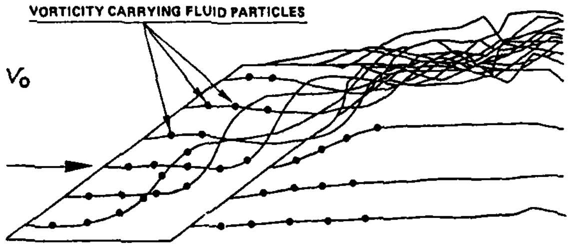

Fig. 88. Schematic of vortex emission lines over a swept wing at high angle of attack

The point vortex emission concept does not solve the problems arising with conventional sheet methods immediately downstream from the boundary layer separation. Nevertheless, it eliminates the arbitrary nature of the wake development in previous methods. This constitutes an appreciable step forward, although it must still be confirmed for helicopter fuselages. However, the increased accuracy compared with conventional vortex sheet methods is obtained at relatively high cost in terms of computer time (on *the order of 30 minutes* CPU time on a CRAY-1 computer for a 700-panel fuselage). On the other hand, this method has a major advantage over every other, namely the absence of any 3-D mesh in the wake.

2.2.4 CONCLUSION ON COMPUTATIONAL FUSELAGE AERODYNAMICS

In the final analysis, it must be acknowledged that the theoretical developments to date in the area of fuselage aerodynamics do not represent any decisive progress over conventional experimental work. They do provide interesting analyses of the airflow that supplement the wind tunnel test results and offer criteria for comparing different geometries, but the *principal objective*, *namely drag prediction*, *remains* beyond reach.

The existing methods are thus unsatisfactory, and fundamental work must continue, as undertaken for example since 1983 by the GARTEUR Action Group on fuselage aerodynamics, a joint research program between WESTLAND Helicopters Ltd, AEROSPATIALE and MBB (with the participation of RAE, ONERA and DFVLR) which *is devoted to the development of theoretical* methods for reducing fuselage drag.

Accurate drag predictions do not appear to be achievable with "lightweight" methods, and it seems illusory to attempt to deal with flow regions as complex as afterbody wakes with the schematic vortex sheet models that were used successfully on fixed wings.

An examination of the airflow around a fuselage reveals distinct regions with specific characteristics requiring the attention of specialists (*e.g.* the 3-D boundary layer). Prior to the final synthesis by the helicopter manufacturer, the assistance of research organizations is required to develop operational computer codes, the building blocks that will then be assembled in a complete program. This work has been successfully accomplished in the last two decades, first for 2-D boundary layer methods, and then for 3-D methods. It must now be continued for the present stumbling block: wake cal*culations.*

ACKNOWLEDGMENTS

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3.3 Sound Due to Blade Forces

3.3.1 Steady Airloads

In a classical early paper on propeller noise, Gutin[32] in 1936 showed that important features of the noise radiation could be explained in terms of the propeller thrust and torque. He represented the propeller disc by a surface of dipoles which were energised by the passage of the rotating blades. The Fourier components of the sound field at position r, θ (θ measured from the thrust axis) were given by

$$c_n = \frac{n\Omega}{2\pi cr}\left\{T_0 \cos\theta - \frac{D_0}{M}\right\} J_n\,(nM \sin\theta) \qquad (20)$$

where n is harmonic number relative to rotational frequency Ω, T_0 and D_0 are the steady blade thrust and drag forces moving circumferentially with Mach Number M and J_n is a Bessel function of the first kind. Although the equation should be summed over a suitable radial distribution of blade loads, reasonable results can be obtained using a single force pair acting at about 80% radius. Note also that for a B-bladed rotor, the sound pattern of each blade is out of phase with those of the other blades so that harmonics which are not multiples of B cancel out. For the remaining harmonics n = mB and T_0 and D_0 are summed over all blades.

Gutin's theory provides a reasonable estimate of the first few harmonics of propeller or rotor noise at least for tip Mach numbers greater than about 0.6 both in terms of level and directivity (Figures 18 and 19). However it seriously underestimates higher

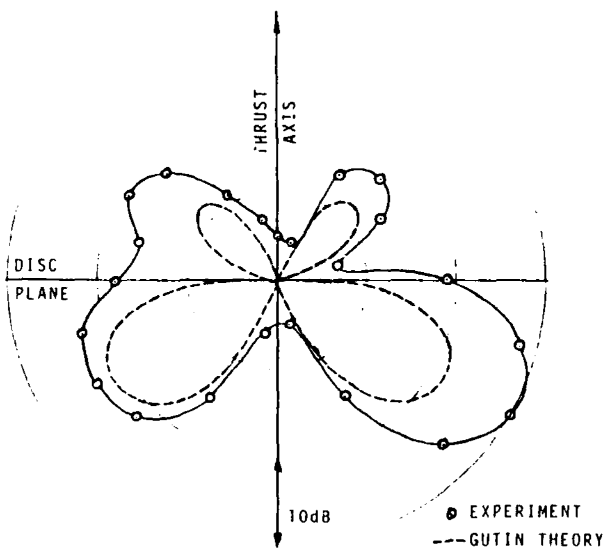


Figure 18 Comparison of Gutin's Theory and Experiment[33] for 2-blade Propeller

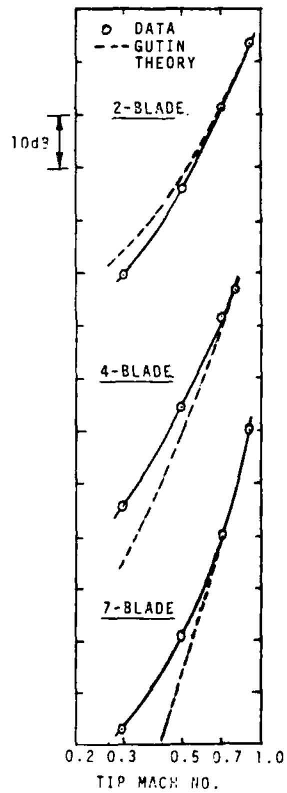


Figure 19 Comparison of Measured and Theoretical Level of First Five Harmonics of Propeller Noise as Function of Tip Speed[34]

harmonic levels, especially at lower tip speeds and this is an especially serious limitation for helicopter main rotors (with blade passing frequencies of 20Hz or less) where harmonics above the 10th or so make the main contributions to subjective annoyance.

3.3.2 Periodic Airloads

The probable importance of unsteady airloads was recognized by many early workers but they were not included explicitly in calculations of rotor noise until some thirty years after Gutin's studies when the problem of helicopter noise was beginning to attract serious attention. Following experimental work in the U.S.A. and England[35,36] in the early 1960's which clearly demonstrated the need for better theoretical understanding, Schlegel

impenetrable surface, all points of which move with the same linear or angular velocity, their result reduces to

$$p(\underline{x},t) = \frac{1}{4\pi}\frac{\partial^2}{\partial x_i \partial x_j}\int_V \left[\frac{T_{ij}}{r|1-M_r|}\right] dV - \frac{1}{4\pi}\frac{\partial}{\partial x_i}\int_S \left[\frac{p_{ij}n_j}{r|1-M_r|}\right] dS(\underline{n}) + \frac{1}{4\pi}\frac{\partial}{\partial t}\int_S \left[\frac{\rho_0 \underline{v}\cdot\underline{n}}{r|1-M_r|}\right] dS(\underline{n}) \qquad (19)$$

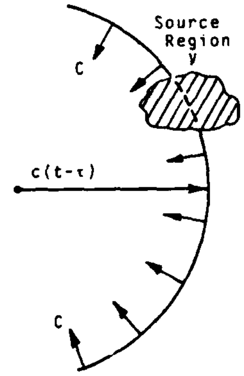


Figure 16 Shrinking Spherical Surface of Constant Retarded Time

where the Doppler effect appears through the $|1-M_r|$ terms. As is the case for the stationary source (18) we see that sound may be generated by (i) Lighthill's quadrupoles in the fluid surrounding the control surface, (ii) the stress exerted across the surface and (iii) by any local volume displacement by the bounding surface (the control surface effectively surrounds a volume of stationary fluid; in order to leave this behind as it moves, the body must exchange mass with its surroundings equivalent to $\rho_0 \underline{v}.\underline{n}$ dS per unit area). The three source terms of (19) are often referred to as stress, force and volume (or mass) sources. All can be of importance to rotor noise generation.

3.2 Rotor Noise Generation

Rotor noise is usually considered to have two distinct but simultaneously occurring components. The first is periodic with a fundamental radian frequency equal to the blade passing rate, $B\Omega$ (number of blades x rotational frequency) and the second is random albeit with an intensity which is modulated at the blade passing rate. Figure 6 illustrates the waveforms and corresponding spectra of typical helicopter sounds which reveal these two types of sound. It is not entirely clear that there is a sharp division between periodic and random components; for example broad peaks in the spectrum lie somewhere between the two descriptions. However the distinction helps in the categorisation of source mechanisms.

The periodic sound is mainly associated with steady blade loads (lift and drag) and volume and due to periodically fluctuating blade loads (due to asymmetric flow, blade/wake interactions, etc.). Time-varying loads which repeat themselves exactly from revolution to revolution give rise to discrete lines in the spectrum (slight variations from true periodicity cause some broadening of these spikes) which can often be identified at very many multiples of blade passing frequency.

The random or broad band component of rotor noise is attributable to turbulence of one form or another. Such turbulence may be generated by the rotor, e.g. in the blade boundary layers, shed wakes and trailing vortices and these sources tend to become important at higher frequencies. However large-scale atmospheric turbulence ingested by the rotor can have a significant effect on noise generation at lower frequencies.

A third component of rotor noise sometimes described separately, is that of impulsive noise or blade slap which could be described as an extreme form of periodic noise since it spectrum contains discrete frequency lines up to very high frequencies (Figures 6 and 7). Its origins can be traced to transonic flow in high-speed rotors and to interactions between blades and concentrated vortices in the wake flow.

Figure 17 shows the breakdown of the various rotor noise sources which is followed in this discussion.

- HELICOPTER NOISE
  - ROTOR NOISE
    - BLADE FORCES
      - PERIODIC
        - Steady Airloads
        - Periodic Airloads
        - Blade-Vortex Interactions
      - RANDOM
        - Vortex Shedding
        - Inflow Turbulence
        - Self-generated Turbulence
        - Tip Vortex
    - THICKNESS
    - FLUID STRESSES
  - OTHER SOURCES

Figure 17 Breakdown of Rotor Noise Sources

monopoles).

These descriptions of the sound fields surrounding the source region have so far avoided the difficult question of how the sound is actually generated, for within the source region the weak perturbation assumptions of the homogeneous wave equation break down. In fact, the exact equation of fluid motion is

$$\frac{1}{c^2}\frac{\partial^2 p}{\partial t^2} - \nabla^2 p = \frac{\partial^2 T_{ij}}{\partial x_i \partial x_j} \qquad (16)$$

where $T_{ij} = \rho v_i v_j + p_{ij} - c^2 \rho \delta_{ij}$

This formulation, derived by Lighthill[30] in 1952, retains the wave equation on the left-hand-side by switching the collection of non-linear terms to the right-hand-side. These include the turbulent Reynolds stresses $\rho v_i v_j$, the viscous stresses $(p_{ij} - p\delta_{ij})$ and $(p - c^2\rho)\delta_{ij}$ which describes non-isentropic effects (δ_{ij} is the Kronecker delta = 1 if i = j; = 0 if i ≠ j).

Thus, in this 'acoustic analogy' the fluid motion may be visualised as an acoustic field (equation(8)) where sound waves propagate with speed c and a source field (equation (16)) comprising a 'quadrupole distribution' of strength T_{ij} per unit volume. The quadrupole nature of the field is revealed by the double divergence on the right hand side of (16) which indicates a double tendency of the sources to cancel. (In the same way that a dipole field is the divergence of a monopole field, so a quadrupole field is the divergence of a dipole field).

The exact solution to (16) is

$$p(\underline{x},t) = \frac{\partial^2}{\partial x_i \partial x_j}\int_V \frac{T_{ij}(\underline{y},\tau)}{4\pi r}\, d^3\underline{y} \qquad (17)$$

Equation (17) can be used directly to estimate the sound radiation from a region of turbulence in which T_{ij} is finite. However rotor noise is due to the interaction between the fluid and the moving blade surfaces and further analysis is necessary to provide a starting point.

Ffowcs-Williams and Hawkings extended the Lighthill formula to include the effects of moving bodies[31]. Their results, which provide the basis for most modern methods of calculating rotor noise, were obtained by defining the right hand side of equation (17), i.e. the Lighthill quadrupole source, as a distribution inside the immersed body and then using the divergence theorem to convert the volume integrals into surface integrals over a surface enclosing the body. The Ffowcs-Williams-Hawkings equation is

$$\begin{aligned} p(x,t) = {} & \frac{1}{4\pi}\frac{\partial^2}{\partial x_i \partial x_j}\int_V \left[\frac{T_{ij}}{r}\right] dV \\ & - \frac{1}{4\pi}\frac{\partial}{\partial x_j}\int_S n_i \left[\frac{\rho v_i v_j + p_{ij}}{r}\right] dS \\ & + \frac{1}{4\pi}\frac{\partial}{\partial t}\int_S \left[\frac{\rho \underline{v}.\underline{n}}{r}\right] dS \end{aligned} \qquad (18)$$

where $\underline{n}$ is the direction cosine of the outward normal to the enclosing surface. The volume integration is carried out over a limited volume outside the surface S where T_{ij} stresses are significant. The second and third terms arise from the conversion from volume to surface integrals and show that the quadrupole field interior to the body is equivalent to a combination of dipole and monopole sources at the surface enclosing the volumes. The dipole strength density is the fluid stress applied to the surface and the monopole strength per unit area is the mass flux crossing the surface (e.g. due to pulsations of the body's volume). The square brackets denote evaluation at source position $\underline{y}$ and retarded time τ.

Equation (18) is an exact expression for the sound radiation in the presence of a foreign body. It is calculated by integrating the source terms at their relevant retarded times. Since all waves travel at speed c it is clear that this integration is effectively carried out over the intersection between the source region and surface of a contracting sphere of radius $c(t-\tau)$ which is centred on the observer (at $\underline{x}$) since this is the locus of points whose emissions arrive simultaneously at the point $\underline{x}$ at time t (see Figure 16).

For a small, fixed source region, the total duration of these emissions is small. However if the source region is moving with an appreciable velocity component towards the observer, the time for which it is intersected by the integration surface can be appreciably increased, as is the spacial extent of the contributing source distribution. This amplification is a manifestation of the well-known Doppler effect.

Ffowcs-Williams and Hawkings applied their theory to the case of a moving body by a transformation to a co-ordinate system $\underset{\sim}{\eta}$ moving with the body. In the case of a rigid,

The homogeneous wave equation which describes the propagation of sound waves in unbounded space may be written

$$\frac{1}{c^2}\frac{\partial^2 p}{\partial t^2} - \nabla^2 p = 0 \tag{8}$$

where p is the acoustic pressure perturbation, c is the speed of sound and

$\nabla^2 = \frac{\partial^2}{\partial x_1{}^2} + \frac{\partial^2}{\partial x_2{}^2} + \frac{\partial^2}{\partial x_3{}^2}$ ($= \frac{\partial^2}{\partial x_i{}^2}$ in tensor notation). This equation is derived using linearisations which are valid in weakly disturbed fluids.

When considering sound generation, a source region V may be defined within which the right hand side is not zero. Within V we may write

$$\frac{1}{c^2}\frac{\partial^2 p}{\partial t^2} - \nabla^2 p = q(\underset{\sim}{x},t) \tag{9}$$

where $q(\underset{\sim}{x},t)$ represents a disturbance at position $\underset{\sim}{x}$. Defining $\underset{\sim}{y}$ as a source co-ordinate, the solution to (9) is

$$p(\underset{\sim}{x},t) = \int_V \frac{q(\underset{\sim}{y},t - |\underset{\sim}{x}-\underset{\sim}{y}|/c)}{4\pi|\underset{\sim}{x}-\underset{\sim}{y}|} d^3\underset{\sim}{y} \tag{10}$$

Here $\underset{\sim}{x}$ is an observer co-ordinate and the result reflects the fact that the sound at $\underset{\sim}{x}$,t was emitted by sources at $\underset{\sim}{y},\tau$ where the "retarded time" $\tau = t - |\underset{\sim}{x} - \underset{\sim}{y}|c$ If the source volume is small compared with the acoustic wavelength, the variation of τ is negligible for a distant observer and (10) reduces to the approximate result

$$p(\underset{\sim}{x},t) = \frac{Q(t - |\underset{\sim}{x}|/c)}{4\pi|\underset{\sim}{x}|} \tag{11}$$

where $|\underset{\sim}{x}| >> |\underset{\sim}{y}|$ and $Q(t) = \int_V q(\underset{\sim}{y},t)\, d^3\underset{\sim}{y}$

In the limit, when the source is concentrated at a point (11) reduces to

$$p(\underset{\sim}{x},t) = \frac{Q(t - r/c)}{4\pi r} \tag{12}$$

where r = distance between observer and source. This source is called a monopole, strength Q(t). The sound field due to a monopole is omnidirectional.

The sound field generated by the source distribution $q(\underset{\sim}{x},t)$ is, according to (10), the the sum of contributions from all elemental sources in the field. Because of cancellations due to the positions of and phase differences between these elements, the sound field could be very weak. Indeed the net sound may only be non-zero because of retarded time differences.

The monopole distribution $q(\underset{\sim}{x},t)$ degenerates into a dipole distribution if q can be expressed as - div $\underset{\sim}{f}$ for a function $\underset{\sim}{f}$ which vanishes outside the source region. In such fields the instantaneous sum of the monopole elements is zero. In this case

$$Q(t) = \int_V q(\underset{\sim}{x},t)\, dV = -\int_V \text{div}\,\underset{\sim}{f}\, dV = -\int_S f_n(\underset{\sim}{x},t)\, dS = 0$$

This integral is over a surface S which encloses V and upon which $\underset{\sim}{f}$ must vanish. In this case the wave equation becomes

$$\frac{1}{c^2}\frac{\partial^2 p}{\partial t^2} - \nabla^2 p = -\frac{\partial}{\partial x_i} f_i(\underset{\sim}{x},t) \tag{13}$$

which has the solution

$$p(\underset{\sim}{x},t) = -\frac{\partial}{\partial x_i}\int_V \frac{f_i(\underset{\sim}{y},\tau)}{4\pi r} d^3\underset{\sim}{y} \tag{14}$$

A point dipole is the limiting case of two opposite monopoles which are brought together for which

$$p(\underset{\sim}{x},t) = -\frac{\partial}{\partial x_i}\left[\frac{F_i(t - r/c)}{4\pi r}\right]$$

In polar co-ordinates, the sound field of a dipole, which is axi-symmetric, is, in the far-field

$$p(r,\theta,t) = \frac{\cos\theta}{4\pi cr}\frac{\partial F}{\partial t} \tag{15}$$

where θ is the angle to the dipole axis. This is a figure-of-eight pattern with zero sound in the plane normal to the axis (due to cancellation between the constituent

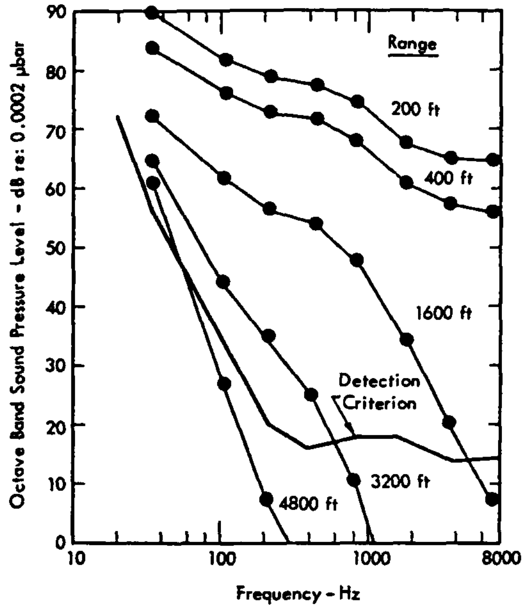


Figure 14 Aural Detectability; Variation of Spectrum with Range[26]

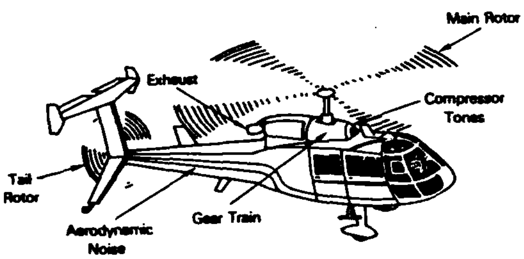


Figure 15 Helicopter Noise Sources

In quiet background noise, helicopters are detectable at distances in excess of eight miles (12km)

2.7 Conclusions

For the present, conventional methods of noise measurement and assessment may be considered adequate to guide the development of improved noise control technology. They show for example that from the viewpoint of both noisiness and detectability it is the mid-frequencies from 100 or 200Hz up to around 1500Hz that pose the main problem.

At the same time, it is clear that there is room for improvement; the scaling procedures are definitely less reliable for rotorcraft noise than they are for fixed-wing noise. An important finding in Ollerhead's study[25] was that the very long attention-arresting sound of an approaching, highly impulsive helicopter did not affect annoyance responses in the laboratory experiments. Yet "hearsay" evidence of complainants near heliports indicates that this may be a particular source of aggravation to people at home. There is therefore a need to investigate the relative roles of detection and loudness/noisiness in community annoyance. This may have an important bearing on certification requirements.

3. NOISE GENERATION

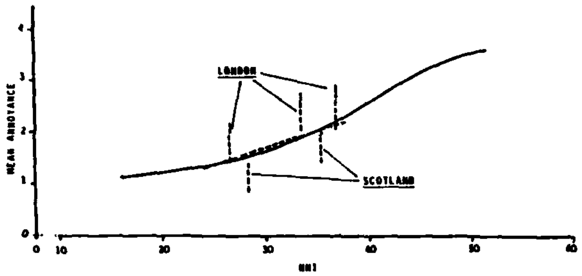
There are many sources of noise on a rotorcraft. Figure 15 illustrates the main ones associated with a conventional turbine-powered helicopter; the main rotor, the tail rotor, the engine, especially its intake (compressor) and exhaust, the gear train and numerous turbulent flows over the vehicle structure (aerodynamic noise). The noise generation mechanisms are complicated by the fact that they may interact with each other. An important example is the strong interaction between the tail rotor and the main rotor wake in forward flight.

Figure 2 shows a typical helicopter noise spectrum which identifies the contributions from some of these sources. It exhibits the multiple spikes of periodic noise at lower frequencies gradually merging into a continuum of random noise at higher frequencies. This spectral presentation has often been used to distinguish between two identifiable types of noise radiated by rotors, termed rotational (periodic) and broadband (random) components. In fact, it is not at all clear that this division aids clear understanding of rotor noise mechanisms; similar ones may underly noise components of each type. Nevertheless the diagram does illustrate clearly the dominant contribution of the rotors to the overall noise and helps to underline the conclusion that noise control effort needs to be directed at the rotors. Unfortunately, the search for a clear understanding of the origins of rotor noise has proved to be surprisingly difficult due to the combined complexities of rotor airflows and aerodynamic noise generation. Considerable effort has gone into many theoretical and experimental studies of rotor noise over the last 25 years or so and these have largely built on extensive foundations laid by researchers investigating the noise problems of fixed-wing aircraft under the broad headings of propellers, jets, compressors and fans.

This section traces some of the main steps in this search and outlines the present state of understanding. It cannot be comprehensive; much of the relevant mathematical analysis is complex and the reader is directed to the referenced works for details. Attention is focussed here on the correlation between theoretical and experimental results, the success of which of course is the real measure of progress.

3.1 Aerodynamic Noise Theory

Detailed treatments of aerodynamic noise theory are readily available, for example in the texts of Goldstein[28], and Dowling and Ffowcs-Williams[29]. A summary of the basic results is presented here by way of introduction to a review of rotor noise research.



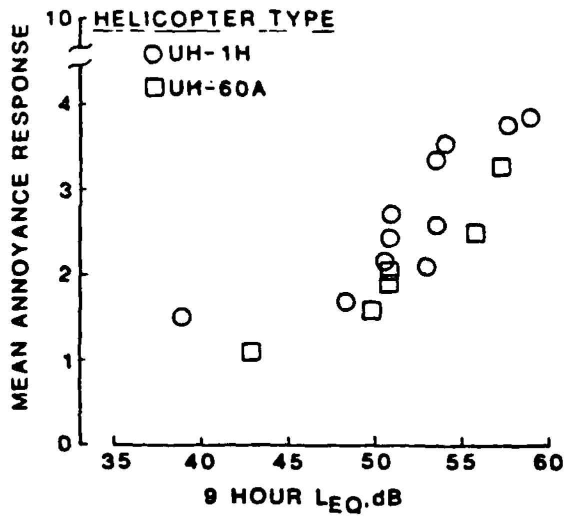


Figure 12 Comparison of Helicopter Annoyance (Vertical Lines Show 95% Confidence Range of Mean) With Trend from Fixed-Wing Surveys. From Atkins et al[22].

Figure 13 Relationship Between Helicopter Noise and Annoyance. Field Data From Powell and Fields[23].

(i) that L_{eq} is a good measure of helicopter noise exposure;

(ii) that when the noise is measured on the L_{eq} scale, impulsiveness is not a significant determinant of annoyance;

(iii) that significant annoyance reactions commence somewhere above 50dB(A) L_{eq}.

The conclusions to be drawn from the work described in this section is that although there is less certainty about the "dose-response" relationships for helicopter noise, as yet there is no compelling evidence to indicate that tolerance thresholds are significantly higher or lower than those for conventional aircraft noise. A suitable planning goal for heliport associated noise in suburban areas is 55dB(A) L_{eq}, measured over a logical measurement period (e.g. 7.00a.m. to 10.00p.m.). Special local considerations, e.g. high background noise or particular noise sensitivity might allow or require this limit to be raised or lowered by perhaps 5dB. In any event such a cumulative noise limit on individual event noise to prevent excessive intrusions when the traffic is low. Although there is no generally accepted standard for such a limit, levels below 80dB(A) SEL are likely to be considered quite acceptable by the great majority of people, even if heard frequently; levels above 90dB(A) SEL will probably give offence to a significant fraction of people, even when heard infrequently.

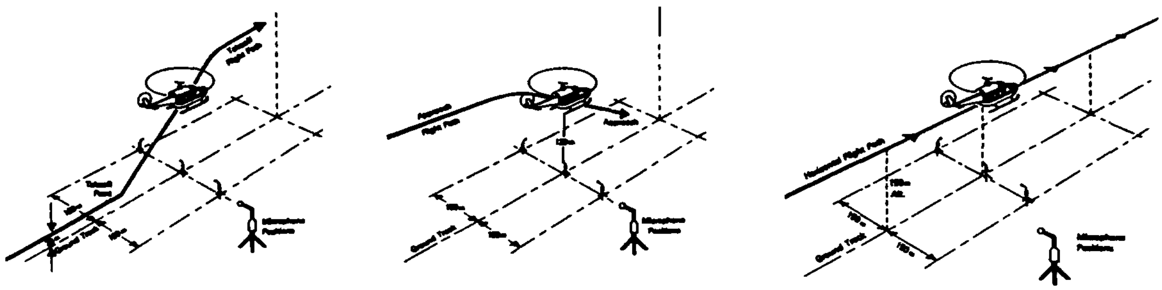
2.7 Aural Detectability

In military applications and to some extent in civil applications too, the major subjective criterion is that of aural detection; i.e. the time for which the approach can be heard. The first analysis was presented by Loewy[24] and an experimental study was subsequently performed by Ollerhead[25].

The three major factors which govern aural detectability of an approaching helicopter are the hearing threshold of the listener, the background noise spectrum and the spectrum of the helicopter sound (the signal). Ollerhead's laboratory studies showed that the signal will probably not be detected if its 'critical band' spectrum is more than 5dB below that of the ambient noise or below the threshold of audibility. (Critical bands are the natural filter bands of the ear - they are approximated at higher frequencies by ⅓-octave bands.) As sound propagates through the atmosphere, high frequency energy tends to be dissipated more rapidly than low so that at typical detection distances the signal tends to have a predominantly low frequency spectrum. But because the audibility threshold also rises very rapidly at low frequencies, the critical frequencies for detection tend to be in mid-range, between 100 and 1500Hz. Figure 14 (from ref. 26) illustrates these points.

Ollerhead's detection model was put to the test in U.S. Army sponsored field studies by Abrahamson[27]. He confirmed the basic accuracy of the model and also devised some significant improvements. Other practical conclusions of importance were

- High altitude approaches (>1500ft.) were audible at 50% greater distances than low altitude ones (<200ft.)
- Helicopter noise is sufficiently distinctive that unprepared or inattentive listeners detect it at the same time as alerted ones



(a) Take-off (b) Approach (c) Flyover

Figure 9 Helicopter Noise Certification Measurements[2]

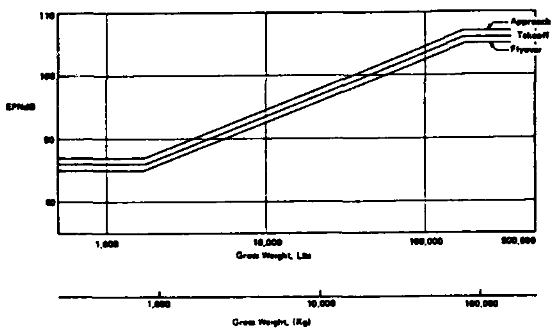


Figure 10 Proposed Helicopter Noise Limits[2]

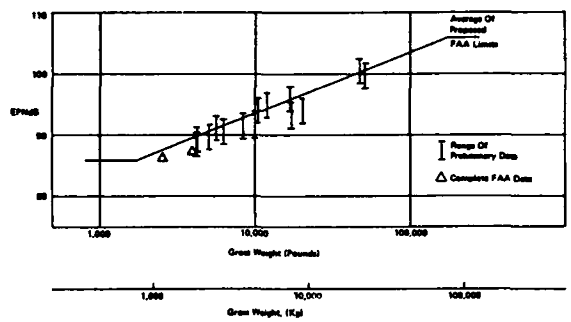


Figure 11 Noise Levels of Current Production Helicopters[20]

The U.S. Government acceded and although a new NPRM is expected shortly, the FAA does not yet require noise certification of helicopters.

Meanwhile ICAO is proposing to raise the limits shown in Figure 10 by 3dB as a temporary measure[21] although this is not expected to be implemented until late 1985.

2.6 Helicopter Noise and the Community

The research discussed previously has revealed no better scale of individual helicopter noise events than SEL or EPNL. We now come to the question of public tolerance; in the community at large is the noise from many helicopters judged differently from the the sound of other aircraft traffic? Evidence from the 10 studies summarised in Table 2 suggests not, but these are based largely on laboratory evaluations of individual sounds.

Two recently published works throw some light on this question. The first[22] was a social survey of public reactions to helicopter noise due to the Heathrow-Gatwick Airlink in London and North Sea operations from Aberdeen Airport in Scotland. Figure 12 summarises the main results of the study which concluded that in Scotland, annoyance due to helicopter operations is of the same order as that due to fixed-wing operations of the same level of noise exposure. In London, helicopter operations are generally felt to be more annoying than those of fixed-wing aircraft but that total annoyance (due to mixed helicopter and fixed-wing traffic) is not inconsistent with total noise exposure as measured by NNI or L_{eq}. It was noted however that variations in the reactions were "somewhat larger than could be expected purely from statistical fluctuations."

The second (Powell and Fields[23]) involved measurements of community reactions to controlled helicopter overflights. Impulsive (UH1) and non-impulsive (UH60A) helicopters were flown between 1 and 32 times per day on 17 separate days over a residential area near a military base in the U.S.A. In the survey, 338 respondents were interviewed to determine their annoyance reactions on a scale from 0 to 10. Figure 13 shows the mean annoyance response as a function of average noise level L_{eq}. The results suggest:

Table 2 summarises Molino's observations. From these he concluded directly that no significant effects of phase relations, tail rotor noise and repetition rate have yet been demonstrated. Although the majority of studies which examined them revealed significant helicopter-fixed wing differences some were positive and some were negative so this effect too was ruled out. Finally, crest-factor, examined in 30 of the 34 studies, was found to be significant in 18 of them. However Molino noted that practically all of the positive results came from experiments involving synthetic (electronically created) sounds while the negative ones were largely based on real or recorded helicopter sounds. Thus crest factor too was dismissed as a factor of independent importance. Molino concluded that "... at present, there is apparently no need to measure helicopter noise any differently from other aircraft noise."

Factor	No. of Studies	Significant? Yes	No	Uncertain
Phase relations	2	-	-	2
Tail-rotor noise	2	-	2	-
Repetition rate	10	4	2	4
Different from fixed-wing	10	6	4	-
Crest-factor	30	18	12	-

Table 2 Factors Studied in 34 Helicopter Noise Perception Experiments Examined by Molino[16]

One of the most recent and largest scale experiments examined by Molino was performed by Ollerhead[19] using 119 test sounds including 89 helicopter and 30 fixed-wing sounds. Major conclusions derived from this study with respect to the merits of EPNL and SEL are:

(i) EPNL and SEL are equivalent in terms of their ability to predict annoyance due to helicopter or fixed-wing aircraft noise;

(ii) Both are very consistent predictors of fixed-wing annoyance; both predict the annoyance levels of helicopters significantly less consistently. This is probably due to the wide range of acoustic characteristics exhibited by different helicopters.

(iii) Impulse corrections did not improve EPNL as a predictor of helicopter noise annoyance. The reason was attributed to the fact that impulsiveness (a) increases the spectral level of helicopter noise in the frequency range 125-500Hz (Figure 8) and (b) causes a significant increase in signal duration which together adequately amplify EPNL or SEL values.

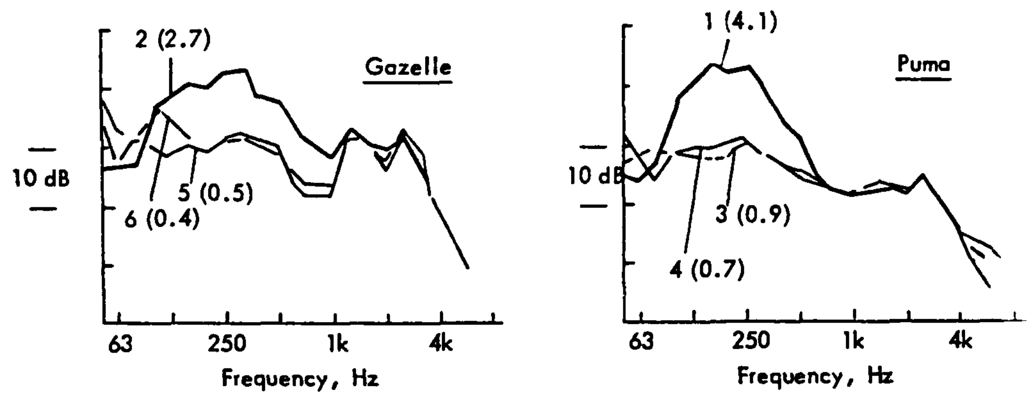
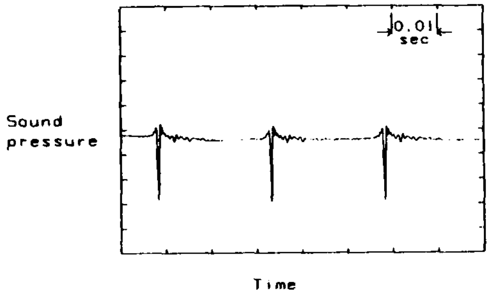


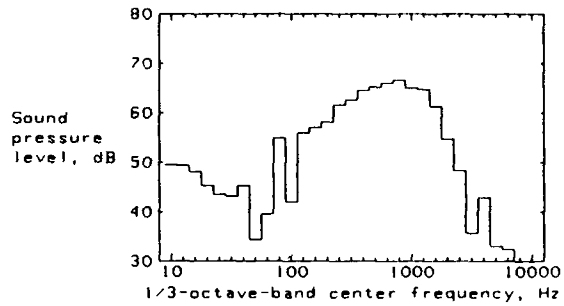
Figure 8 Effect of Approach Blade-slap on ⅓-octave Spectrum of Helicopter Noise. ISO-impulsiveness Corrections are Shown. (Two Non-slap Sounds Included in Each Diagram). From Ollerhead[19].

On the basis of these and other similar findings ICAO decided not to adopt the ISO-proposed impulse correction and eventually framed the helicopter noise certification rule around the conventional EPNL scale[2]. Figure 9 illustrates the noise measurement points during take-off, overflight and approach conditions and Figure 10 shows the proposed maximum allowable noise levels in EPNdB as a function of maximum take-off weight.[20] Figure 11 compares noise levels of current helicopters with the average of these limits[20].

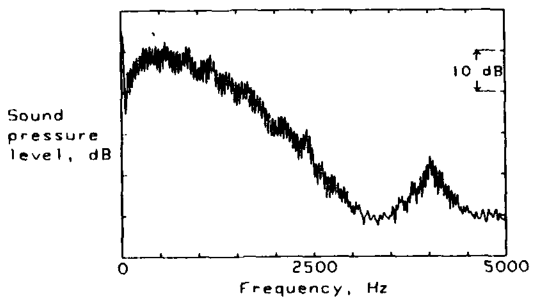
In the U.S.A. the FAA issued a Notice of Proposed Rule Making (NPRM) in 1979,[3] which described the helicopter noise certification rules it proposed to adopt nationally[3]. The noise limits were identical to the ICAO ones but the manner in which they were to be implemented was rather more stringent. For reasons discussed in Section 4.6 the world's major helicopter manufacturers asked for the rule to be shelved pending further research[20].



(a) Time history.



(b) One-third-octave-band spectra.



(c) Narrow-band spectra.

Figure 7 Spectra of Impulsive Sound (Repetition Rate 40Hz), from Powell and McCurdy[13]

working groups to recommend standardised procedures. That eventually recommended by ISO[15] required digitization of the A-weighted sound pressure at a rate of 5000Hz followed by calculation (from the values p_i over short sampling periods) of an impulsiveness parameter I given by

$$I = \frac{\frac{1}{N}\Sigma p_i^{\,4}}{(\frac{1}{N}\Sigma p_i^{\,2})^2} - 1 \tag{6}$$

which is clearly sensitive to the amplitude of the peaks in the time history (usually expressed as a 'crest factor'). An impulsiveness correction Δ was then defined as

$$\Delta = 0.8(\log_{10} I - 3), \quad \text{in dB} \tag{7}$$

with the limitation that $0 < \Delta < 5.5$dB. This is a running correction to be applied to the time history of the measured level; the corrected time history is then integrated to obtain an 'impulse-corrected' exposure level.

Molino[16] has reviewed the findings of 34 psychacoustic studies of helicopter noise perception carried out between 1960 and 1982. In these studies various characteristics of helicopter noise were investigated which may independently influence human perception but may not be accounted for in 'conventional' noise measurement procedures. These were summarized under the following headings (in addition to that of crest factor).

Phase relations: normal spectrum analysis neglects phase relationships between different spectral components of a sound. Although these do not affect total energy, they do affect the shape of the time waveform which may independently influence subjective impressions of the sound.

Repetition rate: although the energy in impulsive rotor noise extends to very high frequencies, the listener is mainly conscious of the pulse repetition rate (which is the rotor blade passing rate). Changes in this rate are very noticeable and it may therefore affect noisiness judgements.

Tail rotor noise: Leverton et al[17] have suggested that helicopter noise judgements may be influenced by the relative contributions of main and tail-rotor noise (which can be clearly distinguished in some cases).

Differences between helicopters and fixed-wing aircraft: it is well known that people react differently to noise from different sources, e.g. trucks, trains and aircraft (e.g. Fields & Walker[18]) probably due to non-acoustical factors such as attitudes towards the source. It is certainly possible that such differences may exist between conventional and rotating wing aircraft.

extent that many airport authorities provide sound insulation grants to residents in higher noise levels.

2.5 Noise Certification

International noise certification procedures are intended to ensure that aircraft are as quiet as technology and economic considerations will allow. The noise must be minimal during sensitive phases of flight and the noise abatement burden should be shared by all aircraft, large and small. The rules therefore tend to specify noise limits covering approach, departure and ground-roll and the limits are related to aircraft size and performance.

The lead in noise certification has usually been taken by the U.S. Government which has also sponsored much research into noise control technology. Existing fixed-wing noise limits have therefore been set such that they can realistically be met (at reasonable cost) by aircraft incorporating the latest technology. There is no evidence that noise certification has significantly hampered aviation and there is no doubt that modern transport aircraft are much quieter than their non-certificated predecessors.

When the time came to include helicopters in the certification process it was immediately felt that the standard fixed-wing noise scales EPNL and SEL would not adequately describe helicopter noise because of its unique characteristics - due not only to different mechanisms but also to very different operating procedures.

In an early study of noise scaling methods Ollerhead[7] noted that

> "On average, the scales were extremely consistent for the piston-engined aircraft sounds but increasingly less so for the jets, the turboprops and the helicopters, in that order.... the results for the helicopters are remarkable in that all the scales are poor. The reason .... is probably related to the domination of the helicopter sounds by low frequency energy of a pulsatile nature ... the subjective effects (of which) require further investigation..."

Of particular concern is the intensely impulsive noise generated by some helicopters in certain flight regimes and commonly known as 'blade slap'. This phenomenon is more readily observed in the sound pressure waveform than it is in the spectrum. Figure 6

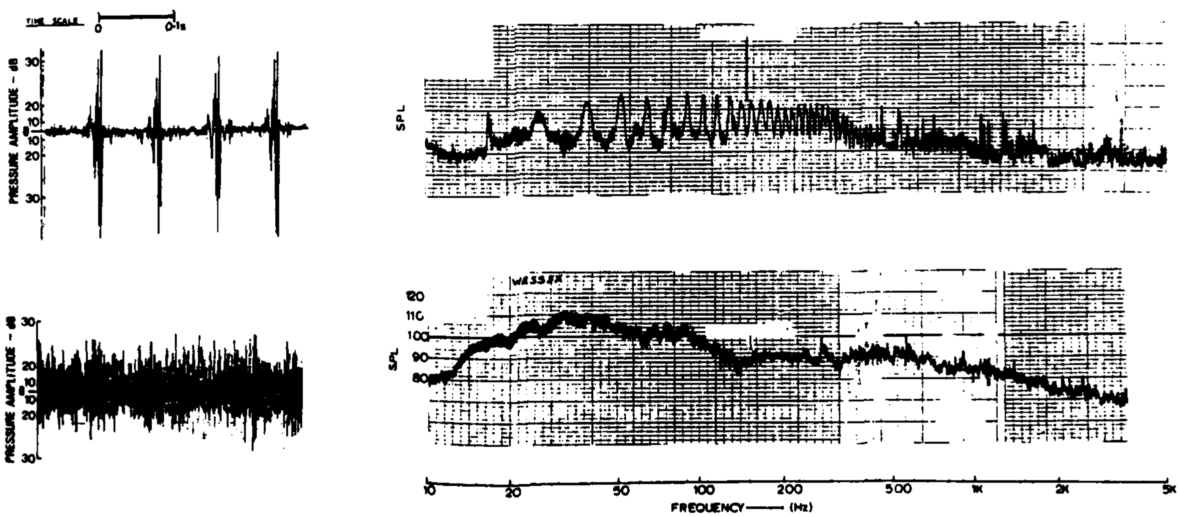


Figure 6 Helicopter Noise Waveforms and Spectra - With and Without Blade Slap (from Leverton[12])

from Leverton[12] compares pressure time histories and spectra with and without blade slap. When slap occurs, most observers agree that it is a particularly intrusive and objectionable sound and audible at great distances. Energy in the pulses is spread over a wide frequency range as the $\frac{1}{3}$-octave analysis of a simulated blade slap (with a repetition rate of 40Hz) in Figure 7 shows.

Many studies have been made of blade slap perception, many of which led to the conclusion that an 'impulsiveness correction' similar to the tone-correction of EPNL, should be applied to L_A or EPNL measurements of helicopter noise. Various methods were proposed (summarized by Berry et al[14]) to quantify this penalty and both ISO and ICAO established

2.3 Cumulative Noise Measures

Noise exposures around airports have to be expressed in terms which sum the effects of multiple events. Many suitable procedures have evolved, the most important of which is called Equivalent Continuous Sound Level L_{eq} given by

$$L_{eq} = 10 \log_{10} \frac{1}{T} \int_0^T 10^{L(t)/10}\, dt \qquad (4)$$

where T is the measurement period and L_{eq} is obviously related to the average sound energy during the period T. As defined by (4) L_{eq} integrates all sound present, including background noise. However the aircraft noise component is easily extracted by using

$$L_{eq} = \overline{SEL} + 10 \log n - 35.6 \qquad (5)$$

where $\overline{SEL}$ is the average aircraft sound exposure level and n is the average number of events per hour. A refined version of L_{eq} used in the U.S.A. for measuring environmental noise levels is 24-hour "day-night sound level" L_{dn} which includes a +10dB weighting for all sound experienced at night (2200 - 0700hrs.). A similar scale based on EPNL rather than SEL is Noise Exposure Forecast (NEF). (In the U.K., aircraft noise is measured on a scale called Noise and Number Index, NNI. The equivalence between NNI and L_{eq} is, approximately 55dB(A)L_{eq} ≡ 35NNI; 70dB(A) ≡ 55NNI.)

2.4 Noise Criteria and Limits

The noise measurement scales have been developed via laboratory, field and social survey research aimed at maximising the correlation between physical dimensions of the noise and human reaction to it (Figure 1). But the selection of specific noise limits is also influenced by economic and technical constraints and they must therefore lie between the desirable and the possible.

The choice of limits is also made difficult by the fact that individuals vary enormously in their reactions to noise. Figure 4 based on data from references 8 and 9 show

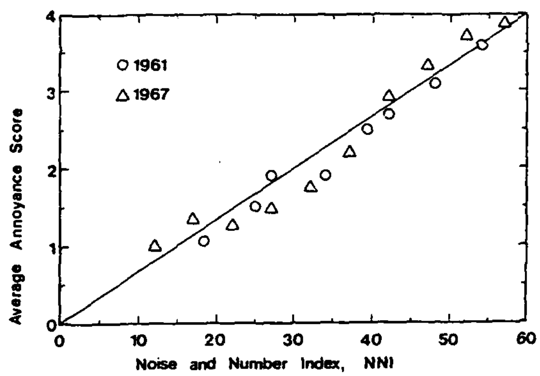


Figure 4 Average Community Annoyance vs. Noise (Data from 8,9)

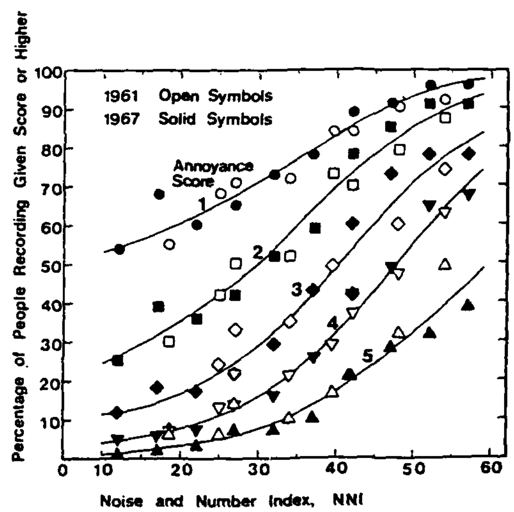


Figure 5 Distribution of Community Annoyance (data from Heathrow surveys[8,9])

average annoyance reactions plotted against noise; the correlation is clearly very high. However, individual reactions are so variable that all shades of opinion can be found at all levels of noise exposure (Figure 5) and there is no clear dividing line between the acceptable and the unacceptable. As noise levels increase, the fraction of people who are highly annoyed increases rather slowly (in round figures about 20% of the total for each 10dB -increase in cumulative noise exposure).

Surveys of aircraft noise annoyance in Britain[8,9,10] and elsewhere[11] strongly suggest that for large and small airfields alike, the threshold of measurable annoyance occurs around 55dB(A) L_{eq}. Some people are annoyed at lower levels but below this threshold the correlation between annoyance and noise is low, suggesting that although noise tends to be identified as the source, people may actually be annoyed more by the presence of aircraft than by the noise they generate.

Above 55dB(A) L_{eq} annoyance grows steadily and levels around 70dB(A) L_{eq} which are reached near big airports are widely recognized as the limits of the acceptable to the

elaborate for specifying noise limits. For this purpose the spectrum is condensed into a single number by a summation process which models that of human hearing. Human frequency analysis is performed in "critical bands" but, very roughly, these can be approximated by $\frac{1}{3}$-octave band analysis in which the audible frequency range is divided into about 30 contiguous bands, each spanning a frequency ratio of $2^{\frac{1}{3}}$ (an octave is a factor of 2). The ear responds differently to sounds in these different bands; higher frequencies are generally louder and noisier than lower ones and detailed measurements of this frequency response have been made and incorporated into various international standards.

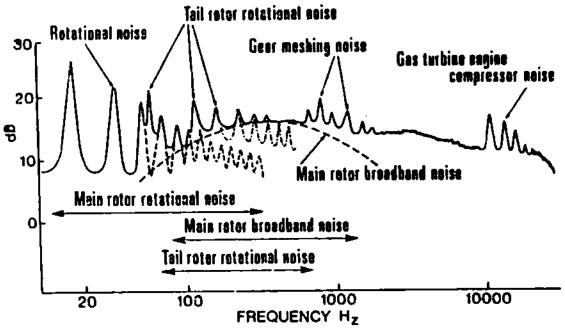


Figure 2 Helicopter Noise Spectrum

Using this standard data it is possible to assign a 'perceived noisiness' value to a particular band of noise with a particular level. The complete set of values for an entire sound can then be summed using a formula which recognises that the noisiest will tend to dominate. The sum is the total perceived noisiness of the sound which is then converted back to a decibel quantity Perceived Noise Level (PNL). The procedure was originally devised by Kryter[5] following earlier work by Stevens[6] (see Ollerhead[7] for a review of its history).

A simpler and much more widely used alternative approach is to measure the entire sound using a single filter (or weighting network) which approximates the frequency response of the ear. The A-weighting illustrated in Figure 3 is used almost universally outside aviation and is certainly the main alternative to PNL for measuring aircraft noise. Despite the fact that A-weighted sound levels (L_A) are much easier to obtain than PNL's, experiments show that with respect to their correlations with subjective judgements, the difference between the two procedures is statistically marginal[7].

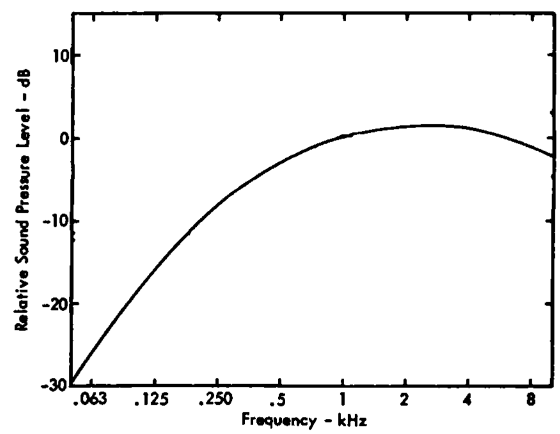


Figure 3 Sound Level Meter A-weighting

Provided levels are measured on such 'subjectively weighted' decibel scales, measured differences and perceived differences are related as shown in Table 1.

Level Difference dB	Weighted Energy Ratio	Difference in Perceived Magnitude
1	1.26	Barely noticeable
3	2	Noticeable
6	4	Clearly noticeable
10	10	Twofold

Table 1 Perceived Differences Between Loudness or Noisiness of Two Sounds

2.2 Single Event Measures

The scales L_A and PNL measure 'instantaneous' sound levels (or rather sound levels averaged over small time intervals, e.g. $\frac{1}{2}$-second). But an aircraft noise event involves a rise and fall of level over many seconds and the time history influences the intrusiveness of the sound. Broadly speaking this depends upon the total sound energy associated with the event $\int I(t)\,dt$ where $I(t)$ is the instantaneous intensity. Although a commonly used alternative is to record the peak level L_{max}, the event is neatly summarised by its Sound Exposure Level given by

$$SEL = 10 \log_{10} \int 10^{L(t)/10}\,dt \qquad (2)$$

where $L(t)$ is usually measured in dB(A).

Perceived Noise Level is time-integrated in a similar fashion but only after it has been 'corrected' for the presence of tones in the spectrum, discrete whines or whistles (normally associated with engine compressors and fans) which strongly influence noisiness judgements. Effective Perceived Noise Level (EPNL) is defined as

$$EPNL = 10 \log_{10} \frac{1}{10} \int 10^{PNL_t(t)/10}\,dt \qquad (3)$$

in units EPNdB. This is the scale used in the noise certification of transport category aircraft. Rules for small propeller-driven aircraft are based on $L_{A_{max}}$[2].

All aircraft are noisy to some degree and most airport operators feel the effects. Subjective aspects of the noise problem have received almost as much attention as physical ones. In many respects they are even more difficult to understand and this too has hampered progress. In civil aviation, noise reductions are brought about partially by legislation, mainly through the process of noise certification[1,2]. For subsonic jet transport aircraft this was introduced fifteen years ago and noise limits have been progressively tightened. The rules were subsequently extended to include propeller-driven, fixed-wing aircraft and rules have been written for helicopters[2,3]. However, at the time of writing no helicopter rule appears to have been implemented.

At the same time the manufacturers are fully aware of the need to reduce noise levels for the comfort of passengers and heliport neighbours (upon which commercial success in the civil sphere ultimately depends) and the discomfort of the military enemy. On the civil side, overland helicopter transport has been surprisingly slow to develop; it is fair to say that to date, it has only been successful over short routes such as airport links where trip demand is high and where alternative (surface) transport is unacceptably slow; due for example to traffic congestion. Until recently lack of progress can probably be blamed on poor helicopter economics but the noise problem is certainly beginning to surface. Britain's only existing scheduled overland helicopter service, the Heathrow-Gatwick Airlink, is shortly to be discontinued on environmental grounds. The City of London helistop has been closed and the country's only existing metropolitan heliport at Battersea operates under a temporary planning agreement. Planning permission for a heliport at the new city of Milton Keynes was recently obtained after a second public inquiry but it is subject to stringent noise limits, which will limit helicopter types and movements. Of particular significance, helicopters will be banned from the imaginative new STOLport to be built in London's dockland.

Thus the fortunes of the rotorcraft industry must be seen to rest to a large extent upon its ability to solve the noise problems. Some of these problems are reviewed in this paper.

2. SUBJECTIVE IMPACT

2.1 Sound Measurement

The role of noise measurement and assessment in aircraft noise control is illustrated in Figure 1. 'Airport noise limits' may be notional goals, i.e. ceilings imposed by the certification process upon the maximum sound levels which could possibly be reached near an airport or they may be real, actual operating limits (on both noise and traffic) which are subject to monitoring by a permanent array of noise measuring terminals. Also, planning authorities require noise guidelines for the control of development around air terminals.

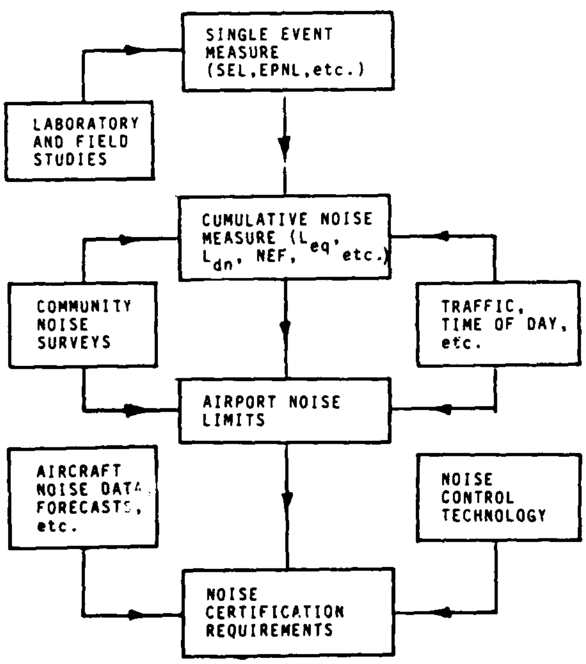


Figure 1 Noise Annoyance Criteria and Certification Standards

Noise certification limits and aircraft noise limits are defined on 'single-event' noise measurement scales. Overall airport noise generation is measured on a scale of cumulative noise exposure which takes account of noise levels, numbers of events and sometimes additional factors like background noise and time of day. Ideally scales used for all purposes should be simply related.

Because the ear has a very large dynamic range and because acoustic stimulus and perception are related by a power law, noise levels are measured in decibels, given by

$$L = 10 \log_{10} (\overline{p}^2) + \text{Constant} \qquad (1)$$

where $\overline{p}^2$ is the mean square acoustic pressure disturbance, proportional to acoustic intensity. The 'overall' level L thus measures the total acoustic energy in a sound although more information is required to evaluate the behaviour and perception of sound which are highly dependent upon frequency. This is partly provided by its spectrum, the distribution of sound energy with respect to frequency in the human hearing range, approximately 20 to 20000Hz. Figure 2 shows a typical spectrum of hovering helicopter noise taken from reference 4. (Note that the frequency scale is also logarithmic.)

Spectral analysis is an important diagnostic tool in acoustic research but it is too

ROTORCRAFT NOISE

by

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SUMMARY

The mechanisms of rotor noise generation are reviewed including methods for noise prediction and low noise design. Attention is focussed on the subjective effects of helicopter noise and the consequent requirements for statutory noise regulation. The economic and operational implications are discussed.

1. INTRODUCTION

The value of the helicopter to modern society is undisputed; military and offshore transport, surveillance and rescue are examples of areas where its role is vital. Unfortunately, one of its major weaknesses is noise. Although rotorcraft noise levels are relatively low by comparison with those of other (larger) aircraft, there are special problems of serious concern.

The first is that the noise is very distinctive; even if the sound level is low its source tends to be unmistakeable and a listener can single out a helicopter from a miscellany of other sounds. This is coupled with the fact that due to current limitations of speed, instrumentation and de-icing, helicopters are usually confined to low altitudes so that en-route noise remains a problem (unlike that of most fixed-wing aircraft). The second is that many helicopters radiate maximum noise in a forward direction; so much so that an approaching helicopter can often be heard for as long as five minutes. From a military or community relations viewpoint this is very damaging. The third problem is that the noise generating mechanisms are extremely complex; to-day, after a quarter-century of serious study, we cannot claim that they are fully understood. A great deal more research is required before manufacturers can be expected to design a rotorcraft which is both quiet and efficient. These problems are considered in this paper. The related problem of high levels of noise and vibration inside the aircraft is not discussed herein.

et al[37] and Loewy and Sutton[38] included unsteady force terms in numerical analyses involving the integration of distributions of point sources making due allowance for retarded time differences. Mainly because of computer time constraints, calculations were restricted to the first few sound harmonics but these showed much better agreement with experimental data than Gutin's theory.

Shortly thereafter Lowson and Ollerhead[39] and Wright[40] were able to extend the analysis to much higher frequencies via closed form solutions to the acoustic equations.

If quadrupole and volume source terms are omitted from equation (19) and the fluid forces are assumed to be concentrated at a single (moving) point, the equation reduces to

$$p(\underline{x},t) = -\frac{1}{4\pi}\frac{\partial}{\partial x_i}\left[\frac{F_i(\tau)}{r|1-Mr|}\right] \tag{21}$$

By use of a suitable transformation to a moving source frame and by expressing the source terms as Fourier series, equation (21) may be integrated to yield the pressure amplitude of the n^{th} sound harmonic

$$p_n = \frac{n\Omega i}{2\pi cr}\sum_{\mu} i^{-\mu}(T_\lambda \cos\theta - \frac{\mu}{nM}D_\lambda)\ J_\mu\ (nM\sin\theta) \tag{22}$$

where $\mu = n - \lambda$ and T_λ, D_λ are the (complex) amplitudes of the λ^{th} harmonics of thrust and drag and θ is the angle between the rotor thrust axis and the observer. Note that when the summation is removed, so that $\lambda = 0$ only (steady load components), equation (22) reduces to the Gutin equation (20).

In equation (22) the λ^{th} loading harmonic is an independent sinusoidal aerodynamic force which completes exactly λ cycles during one revolution of the blade on which it acts. On the axis of the rotor the sound generated by that rotating dipole source is observed to have a frequency λ (since the source-observer distance remains constant). Thus each load harmonic generates sound at its own frequency on the rotor axis. Away from the rotor axis, the sound generated by a particular loading harmonic is modulated in frequency by the Doppler effect of the changing velocity component in the direction of the observer. Fourier analysis would thus reveal that a single load harmonic generates a large number of sound harmonics with a peak amplitude in the vicinity of the basic source frequency. If the source has a large number of (loading) harmonics, each of these generates sound at each harmonic of the blade passage frequency. The effect is illustrated in Figure 20 which shows the calculated contributions of the first 60 loading harmonics to a number of sound harmonics on a four-blade rotor (B=4). Each curve is roughly symmetrical about the mB^{th} loading component (note that due to wave cancellation, the rotor only generates sound harmonics which are integer multiples of the number of blades). More importantly the figure indicates the need to include an adequate range of loading harmonics in any calculation of the noise. For example, significant contributions to the eighth sound harmonic (i.e. 32nd harmonic of rotational frequency) are made by all loading harmonics between the fourteenth and fiftieth. Omission of any of these loads may be expected to result in significant errors and it was shown in ref.39 that the range of interest is roughly

$$n(1 - M\cos\theta) < \lambda < n(1 + M\cos\theta)$$

Figure 20 also shows that the contribution of the $\lambda = 0$ load (steady force) to the higher sound harmonics is very small; for the 4th sound harmonic the steady force contribution is more than 60dB lower than that of the fluctuating forces. Evidently very small load fluctuations can generate high sound levels if their frequencies are high enough.

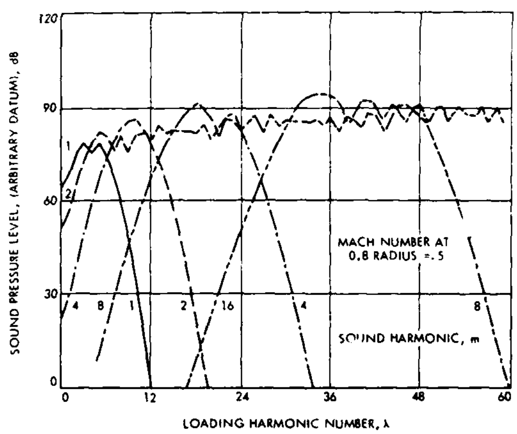


Figure 20 Acoustic Contribution of Loading Harmonics 10° below Rotor Disc

Although the fluctuating force theories explain the deficiencies of earlier attempts to calculate rotor noise they cannot be used to make 'ab initio' noise predictions because they require detailed knowledge of periodic blade load fluctuations. Low-order loading harmonics may be estimated by considering the effects of forward flight, fuselage interference, cyclic blade motions and pitch control but it seems unlikely that the complexities of rotor wake structures will ever be understood sufficiently well to make analytical predictions of the fine-grain detail responsible for the airload harmonics at 200Hz and above, which are important to audibility and annoyance.

To date, realistic predictions of rotor noise have been made by extrapolating measured load data to suitably high frequencies. Lowson and Ollerhead found that loading coefficients (from refs. 41 and 42) decayed approximately as the inverse square of harmonic

number over the first ten harmonics measured (some typical plots are shown in Figure 21); surprisingly little variation of the exponent of this power law was found over a range of advance ratios between 0 and 0.3 and this relationship was used to define the high frequency load harmonics necessary for acoustic calculations, examples of which for both main and tail rotor noise of a light helicopter[26] are shown in Figure 22.

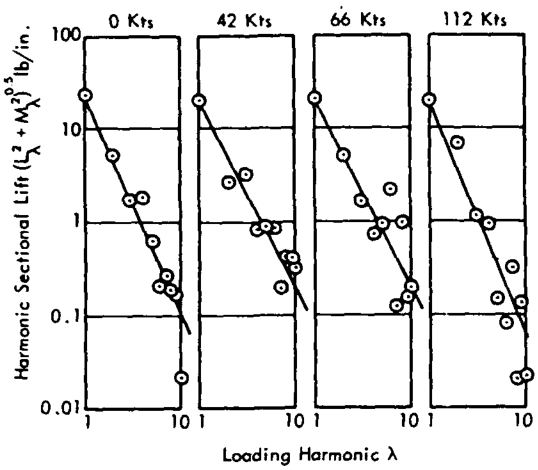


Figure 21 Rotor Loading Harmonic 'Laws' at Various Advance Ratios. Data From Scheiman[42].

At moderate tip speeds, blade loading (dipole) noise tends to dominate rotor noise output and the theory described above, with empirical blade loading data, forms the basis for many practical rotor noise prediction models (see Section 3.5).

From a parameter study using equation (22) Lowson and Ollerhead found that, for periodic loading noise

- the steady blade loads dominate the first few sound harmonics
- higher harmonic sound intensities vary as $(\text{tip speed})^2$
- all sound harmonics are proportional to thrust x disc loading

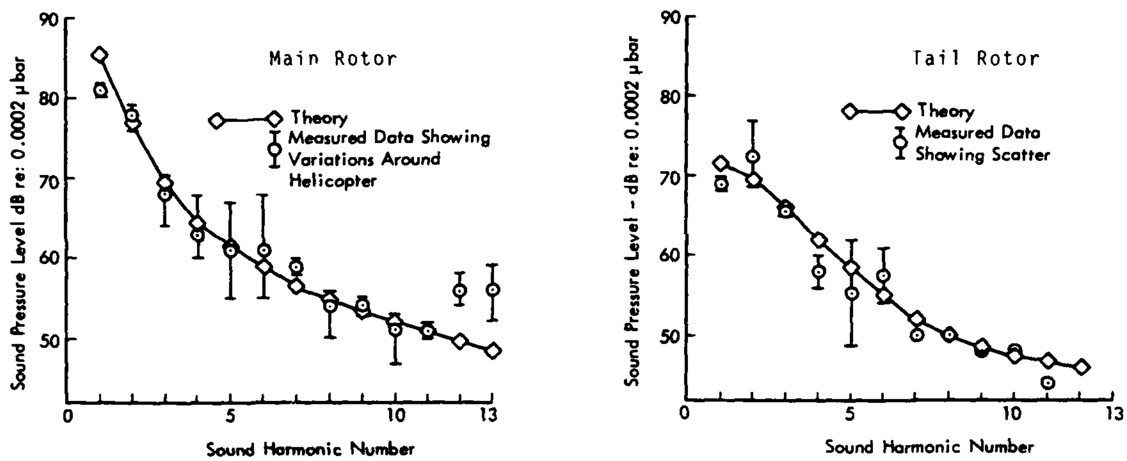


Figure 22 Measured and Predicted Rotational Noise Harmonics for 2-blade Rotors of a Light Helicopter[26]

3.3.3 Blade-Vortex Interactions

The amplitudes of the periodic airloads increase dramatically during certain blade slap conditions and as early as 1972, Cox and Lynn[41] identified an association between blade slap and vortex-induced blade airloads. They concluded that at high flight speeds, the condition was probably due to shock formation on the advancing blade but that when it occurred at low speed it was probably due to interactions between the advancing blade and tip vortices shed by previous blades.

Schlegel et al[37] later observed that blade slap was also caused by stalling of the retreating blade although in tandem rotor machines interactions between the rear rotor and the front rotor wake could be responsible. Simple consideration of wake geometry (Figure 23) indicates that both are possible. Helicopter blade slap due to BVI during low-speed descent is a particularly troublesome noise source to the passengers as well as to people in the terminal area and surrounding community. It tends to prevail at flight speeds between 40 and 80kts. and descent rates between 50 and 100m/min. It has thus been a source of particular concern to researchers.

Leverton and Taylor[43] performed laboratory experiments to investigate the blade/wake interaction problem. They simulated the interaction by generating opposing air jets across the disc plane of a model rotor. By changing the jet positions they simulated both parallel (tandem-rotor) and perpendicular (single rotor) interactions (see Figure 24)

and confirmed this mechanism as a likely source of impulsive noise.

In addition to generating sharp changes in attached flow blade airloads, blade-vortex interactions can also induce unsteady stall and shock-wave formations. Because the associated loading changes occur very rapidly these give rise to strong sound radiation.

Bausch et al[44], like Cox and Lynn, concluded that retreating side interactions are relatively weak due to low blade/observer Mach Numbers. However Hubbard and Leighton[45] measured the relationship between primary (advancing) and secondary (retreating) BVI blade slap using a model rotor in a wind tunnel and found that secondary slap may be of equal or greater intensity than primary slap, especially at positions under the rotor disc where it occurs during shallow angle descent.

Numerous studies both theoretical and experimental have provided good insight into the conditions under which interactions occur(44-50). However, due to the complexity of trailing vortex flows it is not as yet possible to make design predictions of impulsive noise characteristics and levels. The most useful progress in this area is likely to be the use of aerodynamic wake calculation methods to avoid the occurrence of interactions; if such interactions are unavoidable various tip modifications can reduce the intensity of the interaction (see Section 4.2).

3.3.4 Vortex Shedding and Noise

In the earliest studies of propeller noise it was recognised that fluctuating blade pressures were a probable source of acoustic radiation. Indeed their omission was considered to be a possible explanation for lack of agreement between experiment and Gutin's propeller noise theory[34]. Vortex shedding from the trailing edge, similar to the 'Karman street' phenomenon in the wake of circular cylinders at certain Reynolds numbers[51], was postulated as a likely source of blade pressure fluctuations and for three decades or more, the broadband noise radiation from rotors was labelled "vortex noise". Although shed vorticity plays a role in noise generation the Karman street mechanism is relatively unimportant by comparison with many others. However, several studies[52-57] have revealed that airfoils can generate narrowband noise under certain conditions, although this is sometimes referred to as "high frequency broadband noise". (Figure 25 shows data from a model rotor[57].) This phenomenon occurs when the boundary layer on the pressure surface remains laminar to the trailing edge; if this layer is tripped the tone is suppressed.

This periodic vortex shedding appears to be related to instability of the laminar boundary layer but since at normal Reynolds numbers, helicopter main rotor layers are fully turbulent the effect should be of minor importance. However Leverton and Pollard (1973) did observe tone noise on a full-scale rotor operating at low thrust and speed; see Figure 26(a). Full-scale helicopter rotors operating under normal conditions in fact radiate random noise over a wide frequency range; whence the label 'broadband noise' (Figure 26(b)). The following sections describe recent efforts to analyse the underlying mechanisms but this review would not be complete without mention of numerous early empirical 'vortex noise' models, some of which are still used to predict the levels of this important noise component.

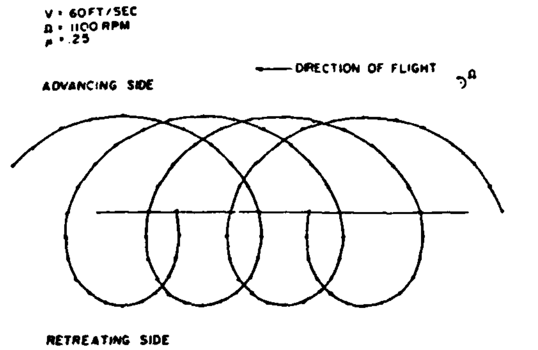


Figure 23 Blade Wake Interactions as Predicted by Rigid Wake

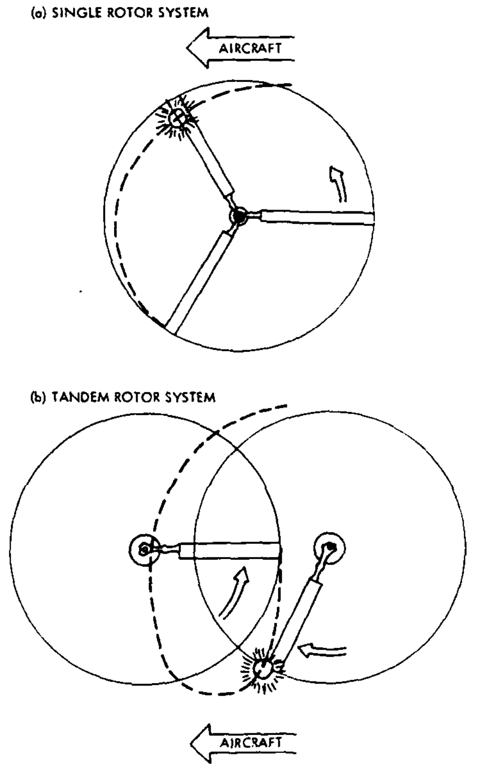


Figure 24 Typical Blade-vortex Interactions for Single and Tandem Rotor Configurations

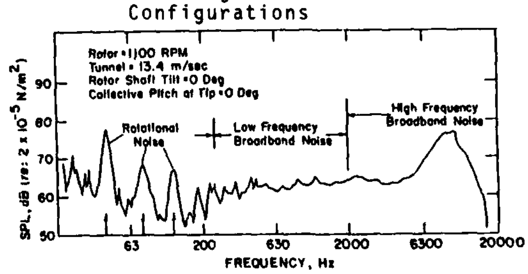
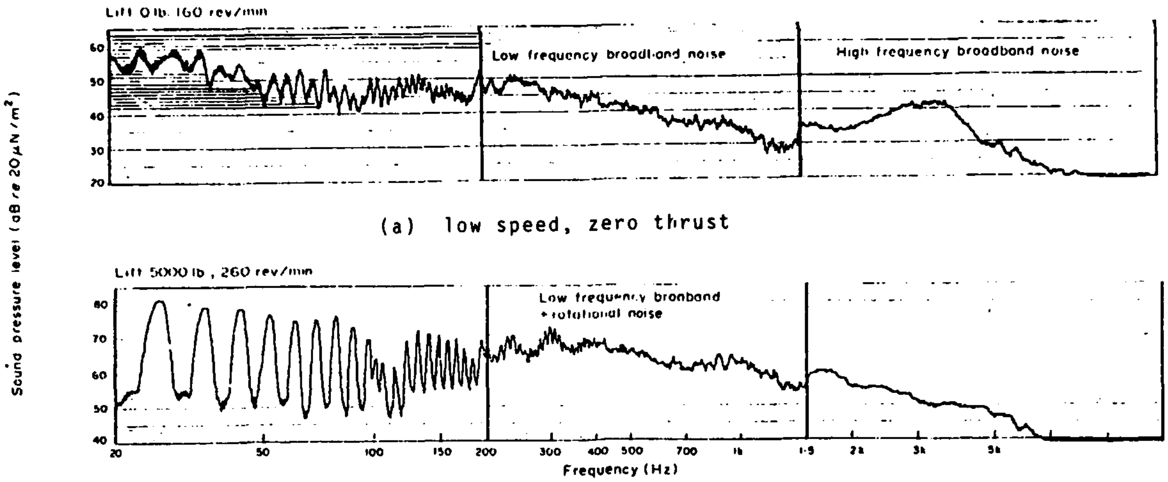


Figure 25 Spectrum of Model Rotor Noise from Aravamudan et al[57]



(a) low speed, zero thrust

(b) high speed, high thrust

Figure 26 Full-scale Rotor Spectra From Leverton and Pollard[53]

Such models have been proposed by Hubbard[57], Davidson and Hargest[36], Stuckey and Goddard[59], Schegel et al[37], Lowson[60] and Widnall[61]. Most of these have been based on the observation (subsequently supported by theory) that high frequency noise output is proportional to (rotor thrust)^2 x blade loading, or (since thrust is proportional to blade area x (tip speed)^2) to V^6S , and also that typical frequency spectra peak at frequencies which scale on a Strouhal Number basis.

Figure 27 shows a collapse of experimental data (mostly from test rigs) presented by Lowson[62], following a format proposed by Widnall (the curved band is from her analysis and shows an upturn at low thrust coefficients caused by blade/wake interaction).

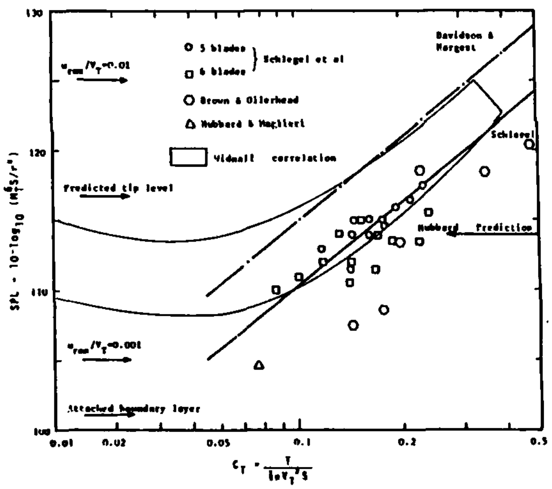


Figure 27 Correlation of Broadband Noise from Lowson[62]

Figure 27 is perhaps more notable for its disparities than its agreement and it does illustrate the kind of uncertainty which has surrounded the broadband noise question.

However, much progress has now been made towards the development of analytical models of various broadband noise mechanisms which are similar to the rotational models described in Section 3.3.2. Again the acoustical theory is relatively well developed; the practical difficulty comes in the specification of the aerodynamic source terms.

3.3.5 Noise Radiation by Rotating Random Forces

An equation for the noise radiated by point random dipoles is devised along similar lines to equation (22). Instead of defining the source airloads as a discrete Fourier series, the blade forces are represented by their spectrum functions

$$\hat{F}(g) = \int_{-\infty}^{\infty} F(t)e^{-2\pi igt}\, dt$$

where $\hat{F}(t)$ is instantaneous force and g is frequency (Hz).

Similarly the spectrum function of the observed sound is

$$\hat{p}(f) = \int_{-\infty}^{\infty} p(t)e^{-2\pi ift}\, dt$$

where p(t) is defined by equation (21). For a rotating source the solution is the series (after Ffowcs-Williams and Hawkings[63]),

$$p(f) = \frac{if}{2cr} \sum_{\mu} \left\{ i^{\mu} \hat{T}(f-\mu f_o)\cos\theta - \frac{\mu c}{2\pi f R} \hat{D}(f-\mu f_o) \right\} J_{\mu}\left(\frac{2\pi f R}{c} \sin\theta\right) \tag{23}$$

where f_o is the rotational frequency $\Omega/2\pi$. This may be seen to have an identical form to equation (22) for harmonic forces using the equivalence $f = n\Omega/2$. However the pressure harmonic p_n in (22) has a direct physical meaning; if phase relationships are ignored, the mean square pressure of the n^{th} harmonic may be expressed

$$\overline{p_n^2} = \left(\frac{n\Omega}{2\pi cr}\right)^2 \sum_{\mu} \left\{ T_\lambda \cos\theta - \frac{\mu}{nM} D_\lambda \right\}^2 J_{\mu}^2(n M \sin\theta) \tag{24}$$

where for B blades, n may be replaced by mB and the forces summed over all blades. The terms T_λ and D_λ are the r.m.s. values of the λ^{th} thrust and drag components. The spectrum function $\hat{p}$ in (23) on the other hand has no direct physical meaning and the power spectral density w(f) must be calculated as

$$w(f) = \lim_{T\to\infty} \frac{1}{T} \int_{\eta}\int_{\eta} \hat{p}(\eta,f)\hat{p}^*(\eta',f)\, d\eta d\eta' \tag{25}$$

where η and η' are independent spanwise co-ordinates and $\hat{p}^*$ is the complex conjugate of $\hat{p}$. This may be evaluated in terms of integrated thrust and drag terms $\hat{T}$ and $\hat{D}$ if it is assumed that they are simply components of the same normal force term $\hat{F}$ (which is in turn defined in terms of a blade load per unit length $\hat{L}$ such that $\hat{F} = \int \hat{L}\, d\eta$). These substitutions lead to the result

$$w(f) = \frac{f^2}{4c^2 r^2} \int_{\eta} \sum_{\mu} w_L(\eta, f-\mu f_o)\ell_{\eta}(\eta, f-\mu f_o) \left| G_{\mu}(\eta, f)\right|^2 d\eta \tag{26}$$

where w_L is the psd of the spanwise loading,
ℓ_η is the spanwise correlation length at η (within which the load fluctuations are phase-coherent)

and

$$G_{\mu}(\eta,f) = i\left(\frac{\hat{T}}{\hat{F}} \cos\theta - \frac{\mu c}{2\pi f_\eta} \frac{\hat{D}}{\hat{F}}\right) J_{\mu}\left(\frac{2\pi f \eta \sin\theta}{c}\right)$$

If the further step is taken of defining a differential pressure psd w_p and a chordwise correlation length ℓ_c such that a surface correlation area $S_c = \ell_\eta \ell_c$ equation (26) reduces to the approximate result

$$w(f) \approx \left(\frac{f}{2cr}\right)^2 \int \sum_{\mu} w_p S_c G^2 dS \tag{27}$$

The problem of using this result lies in the definition of the sourceterm $w_p S_c$.

3.3.6 Inflow Turbulence

Fluctuating blade loads generated by inflow velocity fluctuations are a major source of rotor noise; both periodic and random. At low frequencies the disturbances are mainly periodic in nature. In the mid-frequency 'broadband' range the distinction between periodic and random components is not at all clear. Figure 28 shows Leverton's spectral analysis of the 'broadband noise' generated by a hovering Wessex helicopter in calm and breezy conditions[12]. The light wind causes a pronounced periodicity in this frequency range although the underlying broadband component remains unchanged. The explanation for this is uncertain although it is probably due to the ingestion of elongated eddies, coherent inflow disturbances which are so long that they are cut many times (periodically) by successive blade passages.

Whether or not this particular phenomenon occurs, there are many possible sources of turbulent inflow; turbulent upwash fluctuations, wake recirculation, ambient atmospheric turbulence and passage through the wake from the same or other blades. In forward flight, tail rotors usually ingest the wake of the main rotor which gives rise to additional broadband and periodic noise. Analytical methods to predict rotor noise generation due to turbulent inflow have been developed by Homicz and George[64], Amiet[65], George and Kim[66] and Aravamudan and Harris[67].

Homicz and George[64] treated the general case of turbulence fluctuations and subsonic rotors by solving the Lighthill equation for randomly loaded rotating blading where the loadings were deduced from Dryden's atmospheric turbulence model[68]. This initial work showed that the large scale components of turbulence could explain the 'nearly periodic' but finite bandwidth radiated sound at low frequencies but was unsuited to high frequency analysis due to excessive computer time. This was because with large eddy sizes account had to be taken of inter-blade correlations.

George and Kim[66] therefore used a simpler method for high frequency noise in which, because of smaller eddy sizes, blades could be treated independently. Representing blade loadings by concentrated dipoles, they were able to use the theoretical expression for acoustic spectral density from Ffowcs-Williams and Hawkings[63] (equivalent to equation (27))

$$w(f) = \left(\frac{f}{2cr}\right)^2 \sum_{n} \langle F_r(f-n\Omega)\rangle J_n^2\left(\frac{f\, M \sin\theta}{\Omega}\right) \tag{28}$$

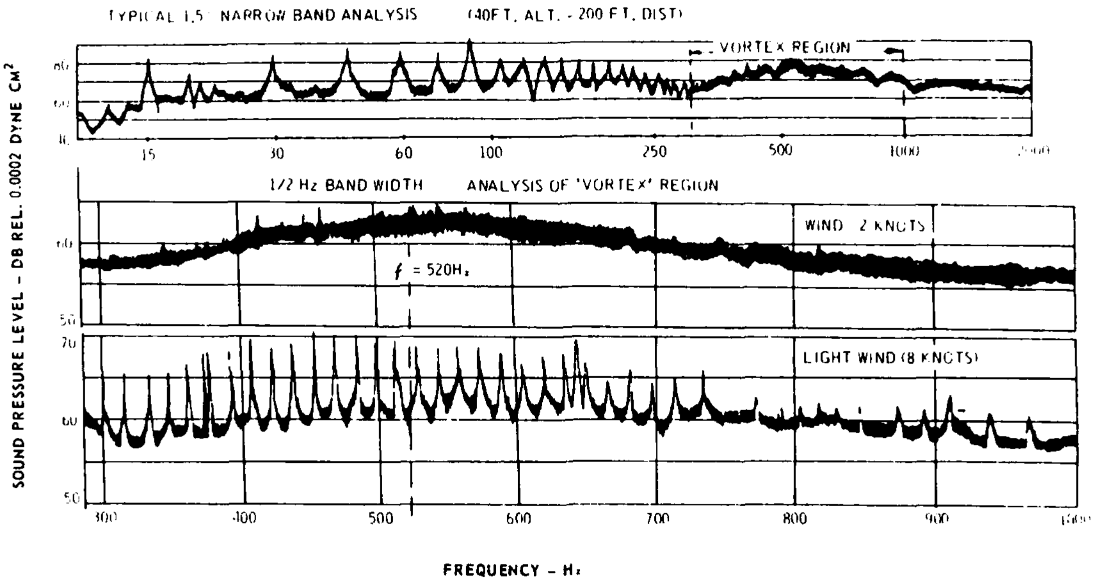


Figure 28 Narrow Band Spectrum of Helicopter (Wessex) Hover Noise. From Leverton[12]

where f is frequency in Hz and $<F_r>$ is the power spectral density of the blade force in the r-direction.

Calculation was further streamlined by including in the summation only those source frequencies of significance to the resulting acoustic frequencies.

The point (compact) source assumption is only exactly applicable if the wavelength is large compared with the blade chord and length. However the authors argued that for rotating sources, the consequent directivity errors would tend to be averaged out. Another problem is that equation (28) applies to statistically stationary force components F_r. This is acceptable for lift (thrust) fluctuations but not for the drag terms which are highly non-stationary (in the observer direction), due to blade rotation. Because the latter were consequently excluded significant errors would be expected near to the rotor plane (where the drag terms predominate).

The lift force spectra were estimated using available atmospheric turbulence models and airfoil lift response functions and various approximations were introduced to reduce the amount of computation involved. Figures 29 and 30 show comparison of theory and

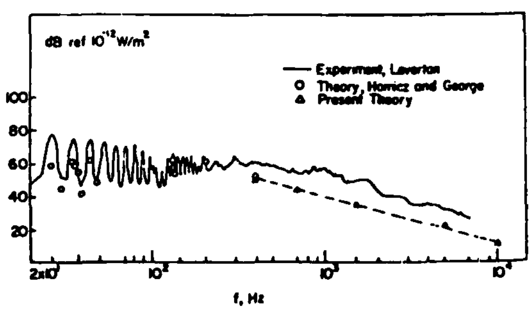


Figure 29 Comparison of Theoretical In-flow Turbulence Noise With Experiment. From George and Kim[66]. Data from Leverton[69]

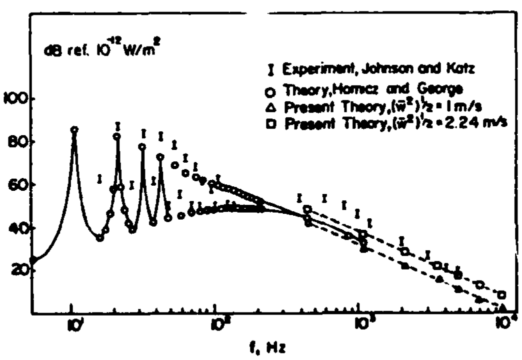


Figure 30 Comparison of Theoretical In-flow Turbulence Noise With Experiment. From George and Kim[66]. Data from Johnson and Katz[70].

experiment taken from George and Kim's paper. Figure 29 is the spectrum measured by Leverton[69] above a full-scale whirl stand upon which the rotor was inverted to eliminate wake recirculation. The low-frequency levels were obtained using the more complete but elaborate theory of Homicz and George. The high-frequency levels were obtained using the approximate high-frequency theory. The same is true in Figure 30 where the experimental data is from Johnson and Katz[70]. In both cases the low frequency agreement is good and the complete and approximate theories agree at the common point at 400Hz. The high-frequency predictions using the 'normal' atmospheric turbulence model are some 10dB low but George and Kim point out that this error could be explained by eddy elongation as the flow contracts upon entry to the rotor disc. This phenomenon is well-known by aero-engine manufacturers who use honeycomb shell flow conditioners to break up these eddies in static engine tests in order to simulate the non-contracting flows encountered in flight. It may well be that very similar noise amplifications occur when helicopter rotors operate near to the ground.

3.3.7 Self-generated Turbulence

The second important source of broadband noise radiated by a rotor blade is the turbulence generated by the blade itself in the boundary layer and tip region flows. When it occurs, local blade stall can substantially increase boundary layer turbulence levels and this may well be a significant contributor to broadband noise radiation.

In the absence of surfaces, typical rotor blade boundary layer turbulence would generate relatively weak quadrupole noise. Furthermore, the effect of a nearby surface on this noise would be small if the surface were large compared with the turbulence scale; an infinite surface simply acts as a passive reflector of the quadrupole sound.* However the picture changes near to the edges of such a surface where the reflection is incomplete and the surface dipoles do not cancel. Thus small or finite surfaces effectively amplify the quadrupole noise by a process sometimes called "edge scattering". This source may be particularly important on airfoils where boundary layer flows are convected past the trailing edge.

Despite many studies, both theoretical and experimental, understanding of this mechanism is incomplete and certain fundamental questions remain unanswered. However, recent studies do provide insight into the likely importance of this mechanism.

Since the problem of trailing edge noise was first investigated by Powell[71] in 1959 various theoretical models have been advanced, notably by Ffowcs-Williams and Hall[72], Chrighton[73], Chandiramani[74], Chase[75], Hayden et al[76], Howe[77] and Amiet[78,79]. Because of different geometries and approximations, direct comparison of these models is difficult but Howe[77] examined their application to half-plane problems using a common system of flow parameters and concluded that at least for low Mach numbers they give identical results. In an extension of the theory, Howe showed that the edge noise is proportional to $L\ell V^5(1-M_0-M_n)$ where L is the wetted span, ℓ the turbulence correlation scale parallel to the edge, V is the characteristic eddy convection velocity, M_0 is the free-stream Mach No. and M_n is the component of the boundary layer Mach No. perpendicular to the edge. This result clearly reflects the dipole nature of the edge noise. A fundamental question which remained unanswered was whether or not the Kutta condition should be applied to the trailing edge model. This could cause noise level prediction differences of up to 10dB in certain cases and a need for experimental clarification was identified.

Trailing edge noise theory has been applied to helicopter rotors by Kim and George[80], Schlinker and Amiet[81], and George and Chou[82]. Kim and George used the simple rotating dipole model given by equation (27) above, estimating the source strength from Amiet's model[78,79] of trailing edge noise radiation from a fixed blade (which in turn requires only blade surface pressure spectra as input). Like the inflow turbulence predictions, these too are acknowledged to be inaccurate at angles within about 15° of the rotor plane due to the neglect of drag forces.

A flat plate blade was assumed for which, following Amiet, the loadings were estimated by accounting for both the convecting pressure pattern and the pressure field induced as the flow adapts to the end of the surface to satisfy the Kutta condition (which was required in this instance).

Figure 31 taken from reference 80 compares calculated trailing edge noise levels with two measured rotor noise spectra. It should be emphasised that these theoretical predictions, unlike the harmonic loading results presented previously, involve no arbitrary or empirical factors. However the calculated trailing-edge noise levels fall below the measured levels in these two cases. Kim and George concluded that the discrepancy may be explained by (a) the significant level of incident turbulence likely in these particular experiments and/or (b) the assumption of the trailing-edge Kutta condition which will lead to lower theoretical noise levels.

3.3.8 Tip Vortex Noise

Another source of rotor broadband noise is associated with tip vortex formation. Lowson et al[83] found that changes in tip shape modified broadband noise output from a

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*It may be argued that by generating fluctuating pressures on the surface, boundary layer turbulence generates dipole sound. This is merely a different viewpoint; due to phase differences the consequent dipole field forms an assembly of quadrupoles - the integrated dipole strength for an infinite plate is zero.

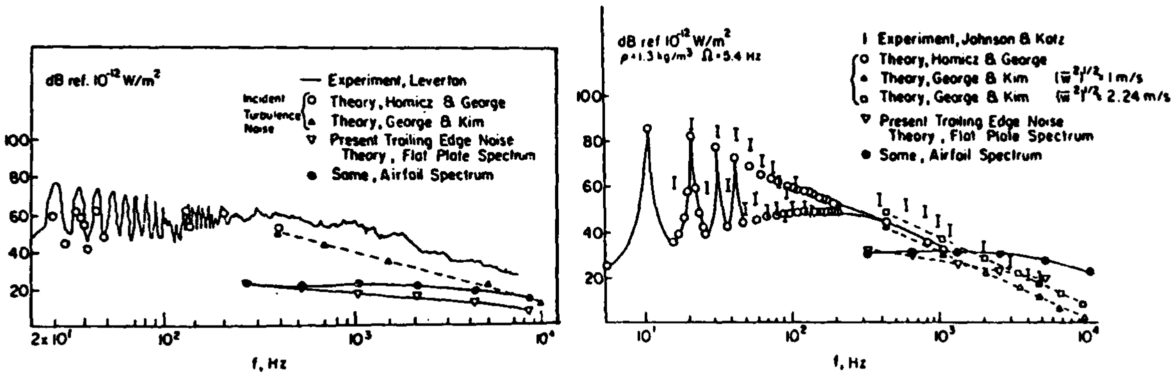


Figure 31 Comparison of Trailing Edge Noise Predictions with Experimental Data. From Kim and George[80]. Data from (a) Leverton[69] and (b) Johnson and Katz[70].

model rotor while Kendall[84], Ahtye et al[85] and Fink and Bailey[86] measured local noise generation from wing and flap tips. George et al[87] identified these sources with the passage of tip vortex turbulence over the trailing edge. Separation occurs on the suction side of the blade near to the tip due to the boundary layer being swept around the tip by the pressure difference. A separated vortex flow results, which is similar to that over a sharp-edged delta wing in subsonic flow, and which generates large surface pressure fluctuations.

3.3.9 Relative Importance of Broadband Sources

In a recently published study, George and Chou[82] compared theoretical predictions of broadband noise from the various sources with experimental data. They programmed for computer solution (a) the turbulent inflow models of George and Kim[66] and Amiet[88], (b) the boundary-layer/trailing edge models of Kim and George[80] and Amiet[78,79], and (c) the tip-vortex noise model of George et al[87].

Figure 32 taken from George and Chou compares the various theoretical predictions with experimental data acquired by Leverton[69] using an inverted, full-scale helicopter rotor on a whirl-tower. This experiment provides what is regarded as the 'cleanest' helicopter rotor data available in that inversion of the rotor minimises the effects of flow recirculation which is normally pronounced when the downwash is deflected by the ground. The measuring microphone was hoisted well above the rotor by a tethered balloon.

Figure 32 corresponds to an angle of 75^{0} 'below' the rotor plane where the theoretical models should lie within the range of their assumptions. It indicates that at lower frequencies, i.e. below 1kHz or so, the noise is dominated by turbulent inflow effects (the inflow turbulence levels were estimated by George and Chou using available atmospheric turbulence models from reference 68) the contributions from self-induced turbulence being many orders of magnitude less. Above 1kHz the latter sources become important with the two theories showing reasonable agreement with each other and Leverton's data (also shown is a boundary layer noise estimate obtained using a semi-empirical prediction after Fink[89]).

Figure 33 relates to wind turbine noise with experimental data from Shepherd and Hubbard[90]. Agreement between theory and experiment is good with trailing edge noise being the dominant source at the high frequency end of the spectrum.

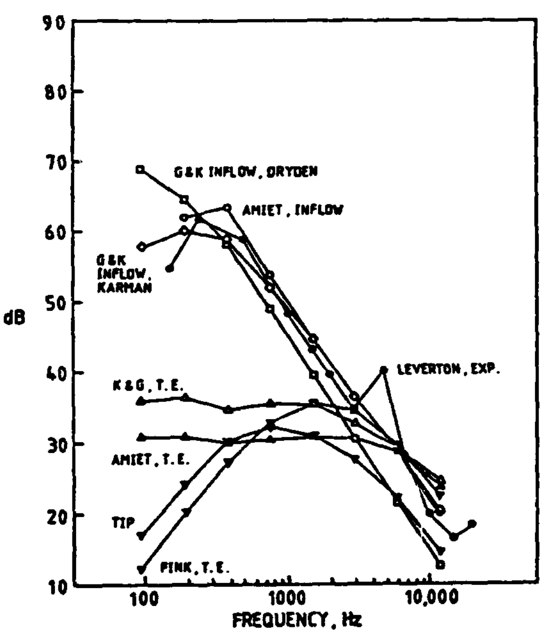


Figure 32 Comparison of Various Theoretical Broadband Noise Components with Experiment. From George and Chou[82] Data from Leverton[69]

Figure 34 shows model rotor data measured by Paterson and Amiet[91] in an anechoic wind tunnel using upstream grids to modify inflow turbulence levels. With no grid (Figure 34a) trailing-edge and tip vortex noise sources appear to be equally important as major noise sources at high frequencies. With a grid (Fig. 34(b)) the inflow turbulence clearly becomes the controlling influence.

3.4 Transonic Rotor Noise; Thickness and Stress Terms

The preceding discussion has concentrated upon force (dipole) mechanisms, which although of major importance at lower speeds is only one of three possible mechanisms, the other two being volume (monopole) and stress (quadrupole) sources. In one of the earliest studies of propeller noise Deming[92] in 1938 considered the possible importance of blade thickness as a volume source. In fact, for 'compact' sources, when the acoustic wavelength is large compared with the source dimensions, as is the case for combinations of small chord, low speed and low frequency, the thickness noise term is very small compared with the force terms. Thickness noise therefore tends to be ignored when considering subsonic rotors. However if the compactness condition breaks down, e.g. due to higher blade dimensions and speeds or smaller wavelengths, thickness noise can be significant.

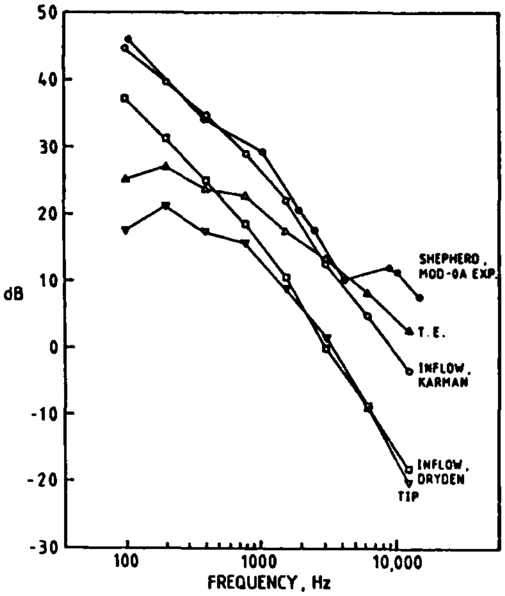


Figure 33 Comparison of Various Theoretical Broadband Noise Components with Experiment. From George and Chou[82]. Data from Shepherd and Hubbard[90]

Lighthill's 1952 general theory of aerodynamic noise demonstrated the importance of fluctuating fluid stresses $\rho u_i u_j$ as sources of sound but it was not until 1969 that Ffowcs-Williams and Hawkings[63] pointed out that these must contribute to rotor noise. If u_i is expressed as $U_i + v_i$ and u_j as $U_j + v_j$ where U and v are mean and fluctuating components, then

$$\rho u_i u_j = U_i U_j + (U_i v_j + U_j v_i) + \rho v_i v_j$$

The last term is the turbulent stress term and the second describes an interaction between mean flow and turbulent velocities. Like thickness noise these terms may become important at high blade speeds. At transonic speeds the $p - c^2\rho$ term in T_{ij} will also require consideration.

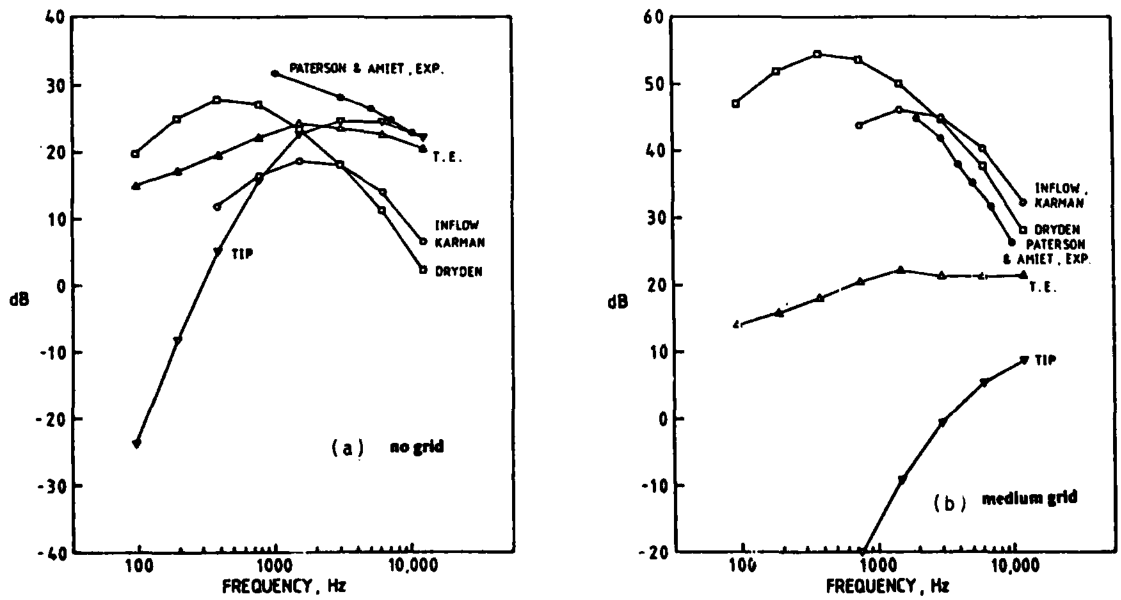


Figure 34 Comparison of Various Theoretical Broadband Noise Components With Experiment. From George and Chou[82]. Data from Paterson and Amiet[91].

A number of theoretical studies of these additional sources have been performed[93-109] and these have shown varying degrees of agreement with experimental data.

Comprehensive measurements of high-speed rotor noise made by Schmitz and his co-workers[110-112] and others have clearly demonstrated the relationships between blade speed and impulsiveness which are naturally very similar to those observed in much earlier measurements of high-speed propeller noise[113-115]. The source is characterised by a highly impulsive signature which is confined to a relatively narrow zone around the plane of rotation. For helicopters this component is radiated forwards and significantly increases the duration of a flyover event (Figure 35). The experiments of Schmitz et al in which measured impulsive rotor noise radiated to a quiet aircraft ahead (Fig. 36)

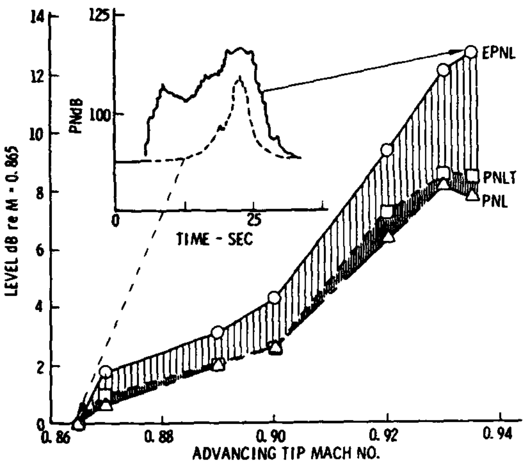


Figure 35 Effect of High Speed Rotor Noise on Duration (through EPNL). From Ref. 116

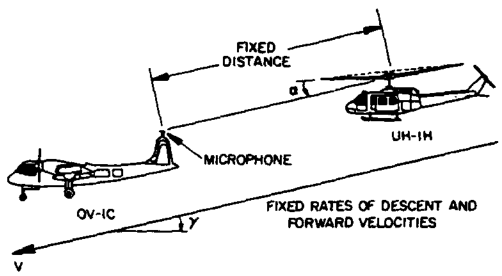


Figure 36 In-flight Far-field Measurement Technique Used by Schmitz and Boxwell[110]

revealed two very important features of high-speed rotor noise. The first is that there is a relatively small range of blade tip Mach number within which there is a rapid increase in the intensity and high-frequency content of the impulses. The second is that the phenomenon is relatively independent of forward flight. Figure 37 from Boxwell, Yu and Schmitz[112] shows the very similar trends in model test results for forward flight and hover.

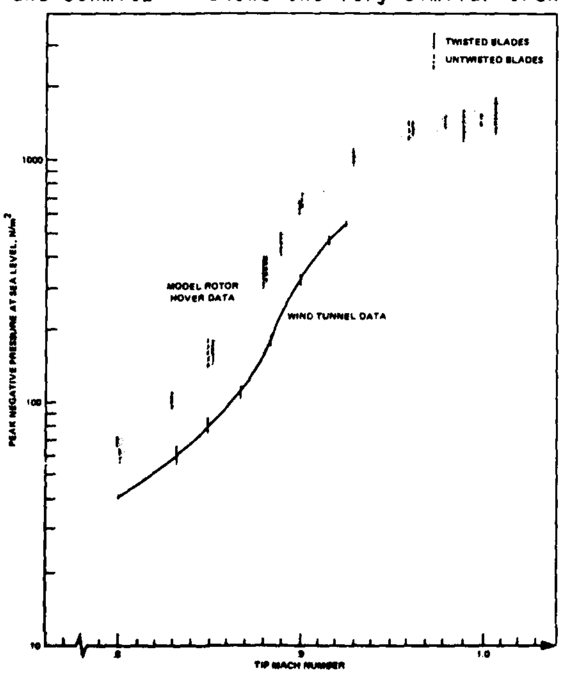


Figure 37 Peak (Negative) Pressure vs. Mach No. for Model in Hover and Forward Flight. From Boxwell et al[112]

The first attempts to model high-speed rotor noise theoretically (e.g. Hawkings and Lowson[93], Isom[94], Farassat and Brown[99], Hanson[100]) concentrated on the volume source in the Ffowcs-Williams-Hawkings equation (i.e. the first term in (19)). Hawkings and Lowson derived a solution which was effectively an extension of the Gutin equation (20) in which a thickness term in $\Omega_0 ch$ is added to the steady force term $T_0\cos\theta - D_0/M$ (h is the blade thickness and the solution involves an integration over the rotor planform area). The characteristic acoustic behaviour of supersonic rotors is contained in the Bessel function $J_n(nM\sin\theta)$ which behaves quite differently depending on whether $M\sin\theta$ is greater or less than 1. For $M\sin\theta < 1$ (which is always true for subsonic rotors) J_n is small and decreases rapidly with increasing n (= harmonic number mB). For $M\sin\theta > 1$, J_n oscillates with an amplitude which decays only slowly with n. Thus for a supersonic rotor, the sound field can be split into two regions: (a) where $M\sin\theta < 1$ and the noise is similar to that of a subsonic rotor and (b) closer to the disc where $M\sin\theta > 1$, where the sound radiation is much stronger and richer in higher harmonics.

At supersonic speeds the sound field becomes a series of weak shocks; Figure 38

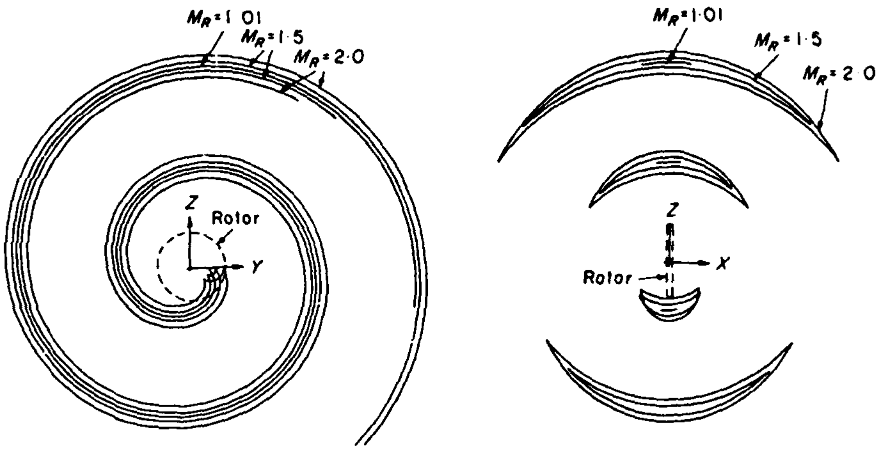


Figure 38 Waveforms for a Supersonic Rotor From Lowson and Jupe[117]

from Lowson and Jupe[117] shows theoretical propagation from a rotor for M = 1.01, 1.5 and 2. Unfortunately, relatively simple monopole theory substantially underestimates the magnitude of the acoustic pressure pulses at transonic Mach Numbers as Figure 39 (from Boxwell et al[112]) shows. The theory also fails to predict an abrupt change in the pressure waveform which occurs at about M = 0.88. At lower Mach Numbers the pulse is effectively symmetrical and its width decreases as speed increases. At higher M the pulse has a non-symmetric 'sawtooth' form with a gradual decrease to the peak under pressure followed by a shock-wave rise to a positive peak (Figure 40).

Hanson and Fink[118], Hawkings[119], Schmitz and Yu[101] and Morgan[107] amongst others have therefore examined the role of quadrupole sources at transonic speeds. For moderately subsonic or fully supersonic flow, quadrupole sources are negligible (due to phase cancellation between the component sources) by comparison with the volume sources. However at transonic speeds, when the retarded time varies little over considerable source volumes, quadrupole sources become equally important (the quadrupoles effectively degenerate into simple sources), the dominant one being ρu^2 where u is the disturbance velocity component in the direction of blade motion. Hanson and Fink[118] integrated the quadrupole source strength around the blade for a transonic flow field calculated using the theory of Spreiter and Alksne[120] to show that the maximum quadrupole contribution is approximately 6dB above the thickness noise for a transonic propeller.

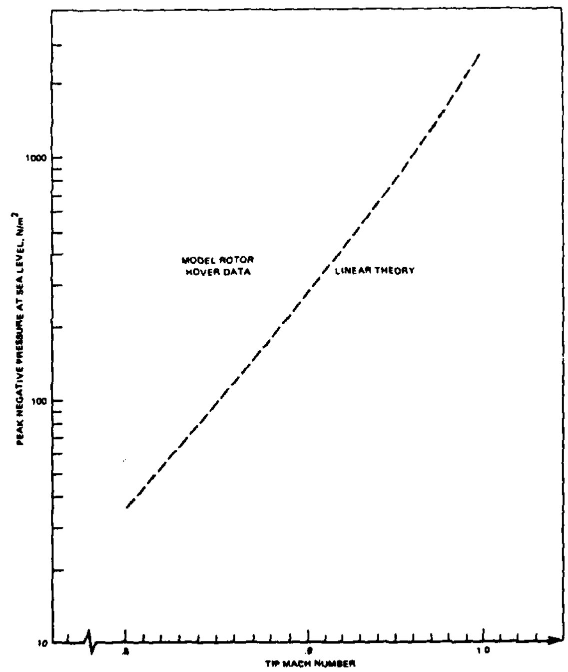


Figure 39 Comparison of Simple Monopole Theory with Model Rotor Data. From Boxwell et al[112]

However, Hawkings[119] pointed out that inclusion of quadrupole effects using the Lighthill acoustic analogy is not straightforward. This is because the source terms depend on flow quantities in an extended region around the blade which, in transonic flow, are significantly modified by the unknown acoustic field; i.e. the 'wave' and 'source' terms in the equation, of fluid motion cannot easily be separated in the Lighthill manner. He went on to show that numerical solutions to a non-linear flow equation developed by Caradonna and Isom[121] gave a better explanation of observed experimental features. Subsequently Morgan[107] extended this work by developing a transformation to an "equivalent acoustic blade" for which the linear equations of acoustics hold and which offers some promise for further theoretical studies of the quadrupole effects.

Schmitz and Yu[103-105] also devised a quadrupole model which had shown good agreement with experimental data at tip Mach numbers up to 0.9 although Aggarwal[109] has noted that their solutions are very sensitive to the integration regions selected and that the apparently promising predicted pressure-time curves may be 'computationally unstable'.

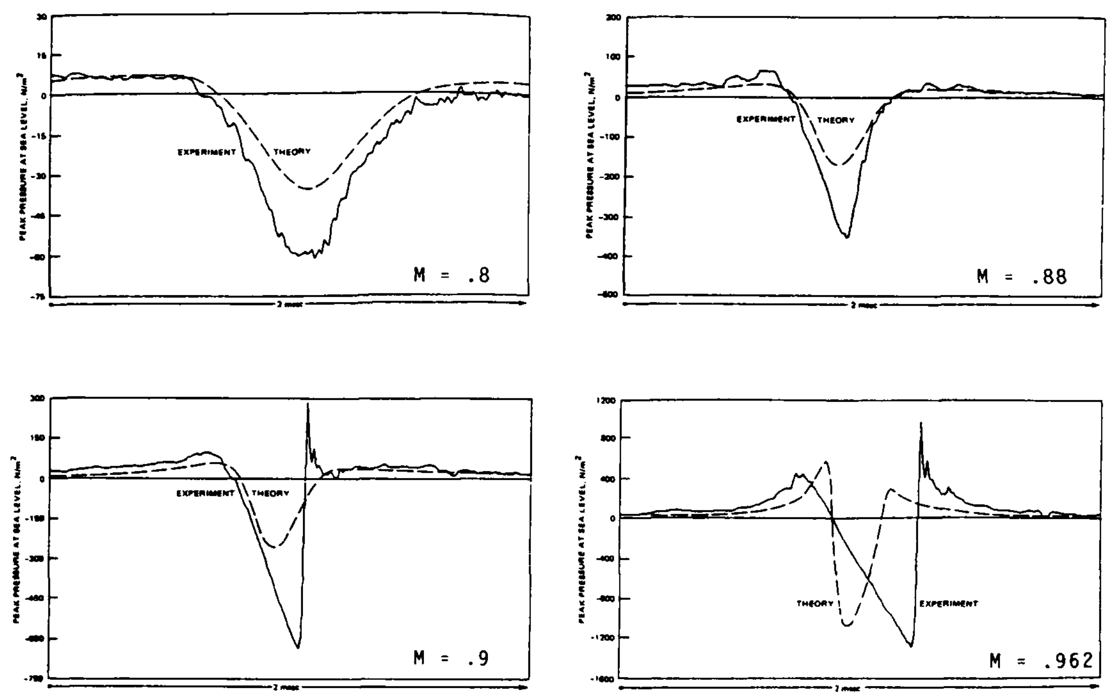


Figure 40 Comparison of Measured and Theoretical Pressure Impulses. From Boxwell et al[112]

3.5 Status of Practical Rotor Noise Prediction Methodology

Although the ongoing research described in this section is making steady progress in the search for better understanding of rotor noise mechanisms it is clear that many are insufficiently understood to be modelled adequately in an 'ab initio' design predition model. Accordingly, the helicopter industry continues to use empirical and semi-empirical models, most of which rely on some form of the Lowson-Ollerhead[39] or Wright[40] rotational noise theories in conjunction with empirically derived airload spectra. Broadband noise prediction is based on the purely empirical methods of Widnall[46], Schlegel et al[37] or Davidson and Hargest[36] and some procedures include account of thickness noise (e.g. based on Hawkings and Lowson[93]) and blade vortex interaction (based on Wright[122]). Reviews of example procedures may be found in Magliozzi[123,124] and Pegg[125].

The Society of Automotive Engineers tested current rotor noise prediction models by asking several companies to calculate certification test noise levels of several helicopters for which experimental data were available. Figure 41 shows the errors in EPNdB. That such methods become less reliable as blade Mach numbers increase is clear from Figure 42 which shows measured and predicted flyover

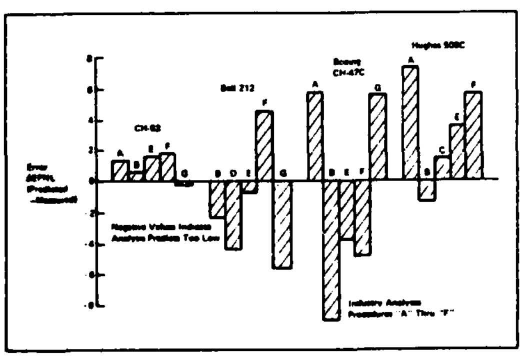


Figure 41 Variation of Helicopter Noise Levels Estimated by Current Semi-Empirical Prediction Models. From reference 20

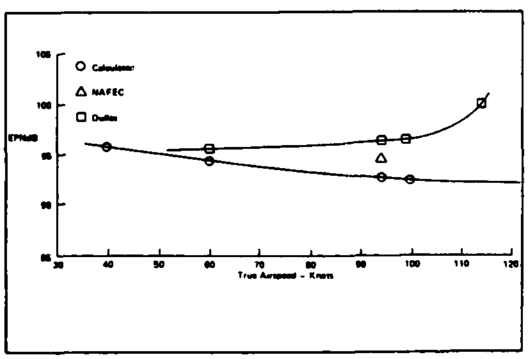
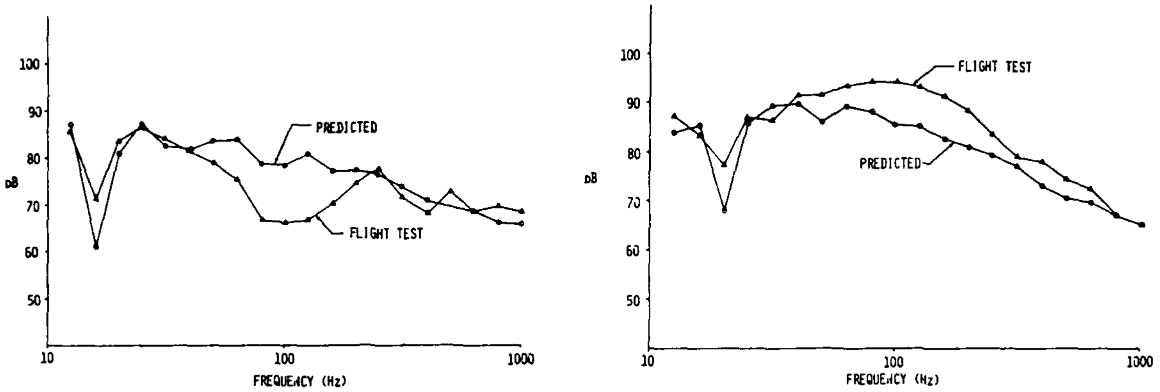


Figure 42 Effect of Airspeed On EPNL Prediction Error. From Reference 20

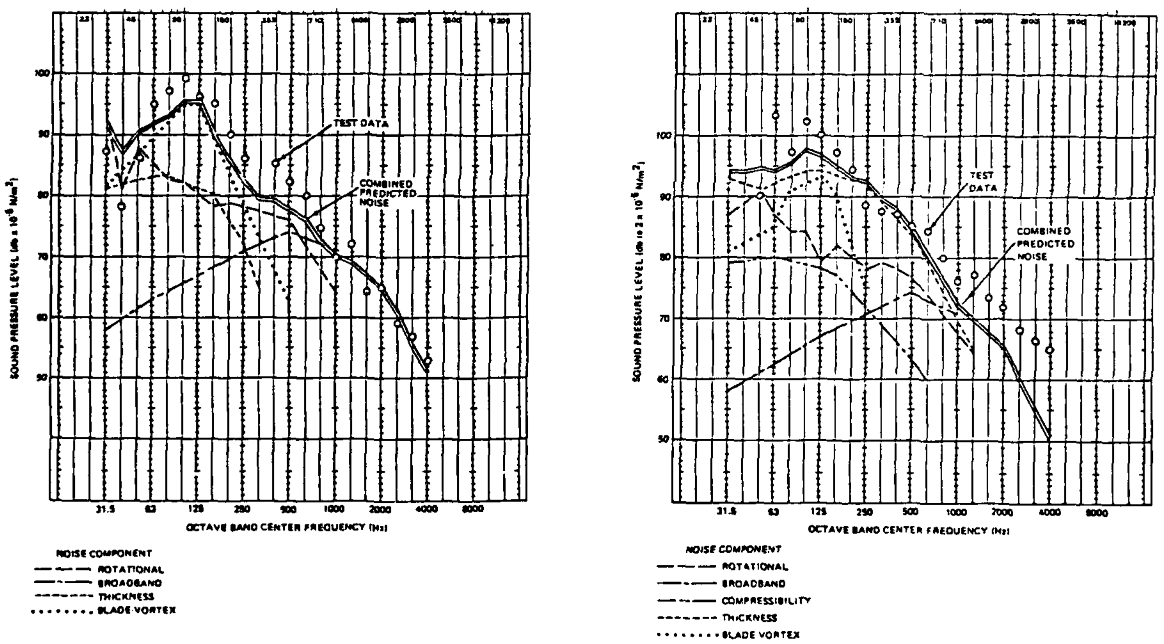
noise levels as a function of airspeed (Bell[212]).

Figure 43 from Gupta[126] and Figure 44 from Spencer and Sternfeld[127] show some com-



(a) 60 knots (b) 165 knots

Figure 43 AH-19 Helicopter Noise Spectra; Measured and Predicted. From Gupta[126]



(a) 141 knots (b) 157 knots

Figure 44 CH-47C Helicopter Noise Spectra; Measured and Predicted. From Spencer and Sternfeld[127]

parisons of measured and calculated noise spectra. The agreement here is obviously quite good but it must be recognised that the applications involve current generation machines for which empirical input data is available or can readily be inferred.

4. NOISE CONTROL

4.1 Requirements

Figure 45 indicates noise level trends of current turbine-powered helicopters by comparison with levels of other aircraft. Figure 46 shows the ground noise 'footprints'

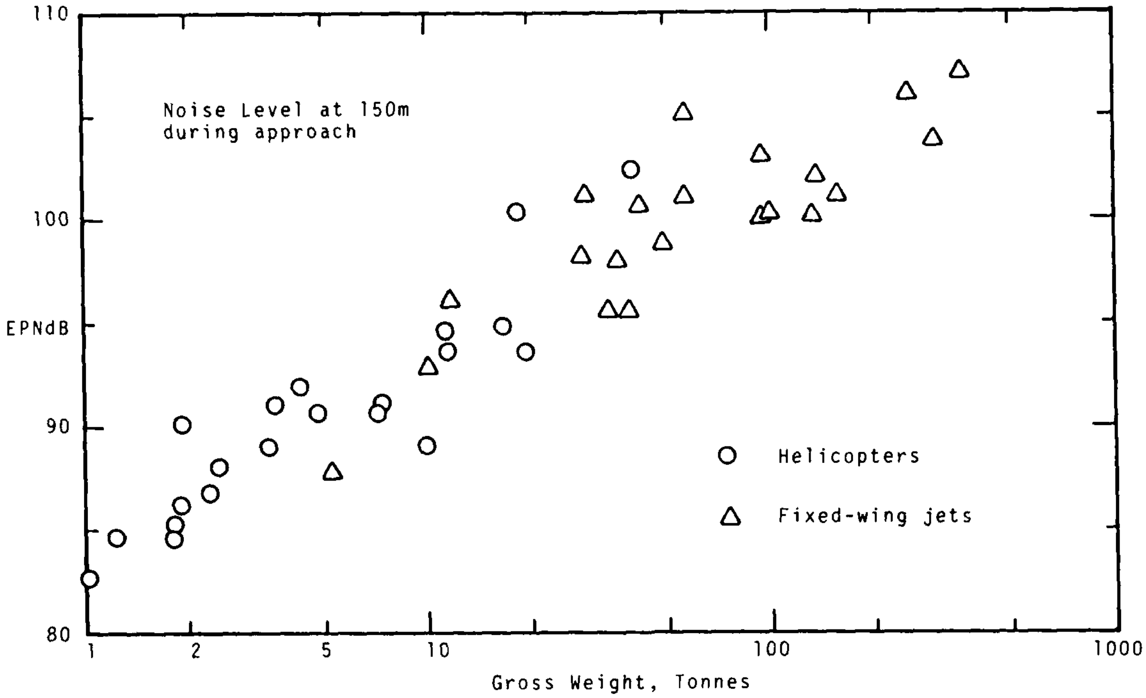


Figure 45 Comparison of Helicopter Noise Levels with Those of Other Aircraft Types

caused by typical helicopter operations from a hypothetical heliport in Regents Park, London. The contours are lines of equal $\overline{SEL}$ averaged for a typical 70% westerly/30% easterly split of operating direction. Allowance has also been made for 60 seconds of hover-taxying to accompany each operation. The contours are labelled for a typical 5000kg commuter transport helicopter with around 15 seats. Levels for a 2000kg executive/air taxi machine with around 5 seats would be approximately 5dB lower. The accompanying table relates $\overline{SEL}$ and L_{eq} as a function of hourly number of operations. The total areas within each contour are listed and Table 3 lists some typical noise levels from other sources to put the SEL values into context.

Bearing in mind that helicopter noise levels greater than 90dB(A)SEL or 55dB(A)L_{eq} may be considered excessive for residential areas* (Section 2) it is quite evident that noise levels of current rotorcraft represent a serious impediment to their use for regular interurban transport and that significant noise reductions, of order 10dB are required if rotorcraft are to be successful in this sphere.

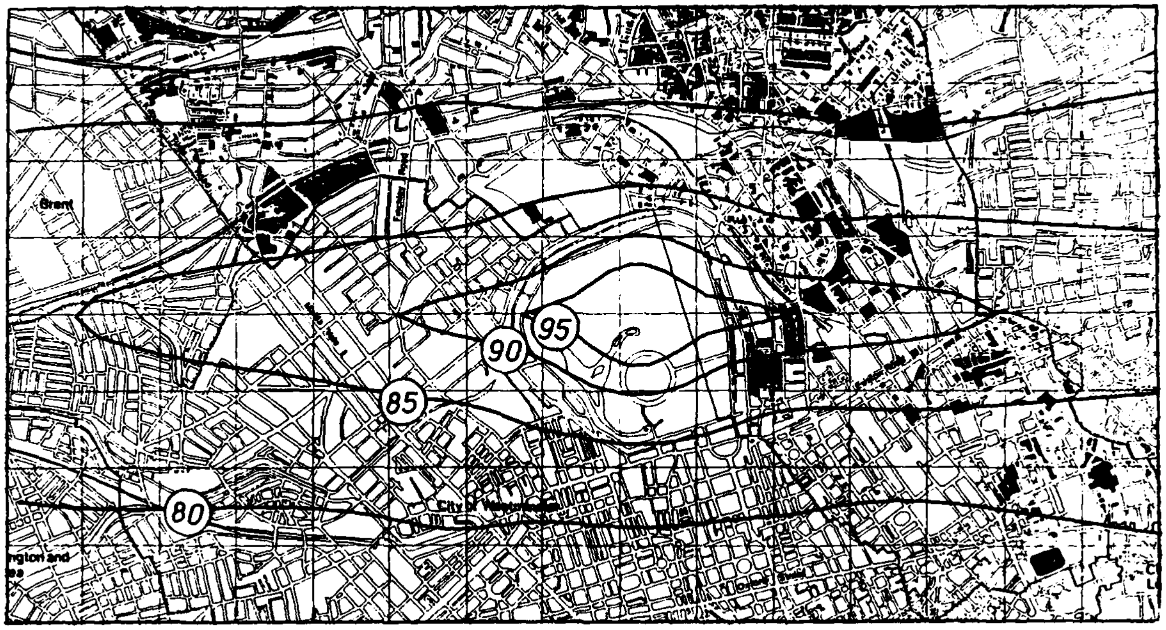
4.2 Practical Noise Reduction

Rotors generate noise by a variety of mechanisms and to reduce the noise output of any particular rotor it would be necessary to know both their relative importance and their dependence upon operating parameters. However there is no doubt that the primary factor is that of rotor speed; multipole noise generation increases with a high power of tip speed. Reduced tip speed (a) reduces rotational noise due to slower source motions, (b) reduces random noise by reducing velocity and surface pressure fluctuation, and (c) reduces high Mach number effects. Unfortunately it also reduces performance.

The low frequency components of main rotor noise are largely controlled by atmospheric turbulence as well as tip speed and are not therefore easily influenced by detail design changes. On the other hand, the mid-frequencies, from perhaps 200 to 2000Hz, which are much more important to subjective perception, are sensitive to rotor-generated turbulence and rotor wake interactions which can be influenced by changes to the geometry and layout of the rotor systems. This is particularly true of the tail rotor which operates in the extremely unsteady environment trailed by the main rotor.

Without doubt the main practical problem is that of blade slap. The high-speed version, which will inevitably remain a problem with the edgewise operation of the conventional helicopter rotors, can be minimised by the use of improved, thinner airfoil sections

*Although in otherwise noisy urban areas limits some 5dB higher might be considered acceptable.



L_{eq} as function of $\overline{SEL}$ and hourly operations*, dB(A)				
Average $\overline{SEL}$, dB(A)	Enclosed Area, Ha (approx.)	Average Operations Per Hour		
		1.1	3.6	11
95	50	60	65	70
90	200	55	60	65
85	>850	50	55	60
80	>1900	45	50	55

*1 operation = 1 arrival + 1 departure

Figure 46 Noise Footprints Around Hypothetical Heliport in Regents Park, London. Noise Levels Are Average Sound Exposure Level Per Operation of Typical 15-seat Transport (subtract 5dB for 5-seat helicopter)

Medium jet airliner, 2 miles from landing Heavy truck at 7.5m, accelerating Express passenger train at speed, 100m	95
Bus at 7.5m, accelerating	90
Heavy truck at 7.5m, 50km/h Suburban train at 100m Twin-prop business aircraft at 1500ft.	85
Bus at 7.5m, 50km/h	80
Automobile at 7.5m, 65km/h	75

Table 3 Typical Sound Exposure Levels (SEL), dB(A)

in the tip region. Improving theories of thickness and quadrupole noise aided by better experimental data from flight and model tests should assist designers in this regard.

Blade-vortex-interaction slap can be reduced by minimising interactions, i.e. by increasing blade-vortex separations, by reducing relative velocities and by diffusing the vortex cores. Figure 47 from White[128] illustrates a variety of rotor tip configurations aimed at achieving this end (see White[128] for publications list). While some of these modifications had some apparent beneficial effect it was found that performance penalties tended to negate them.

The two most promising techniques were an active tip air mass injection system (TAMI) and a (passive) tip described as an ogee. Figure 48 shows the benefits of TAMI, scaled from model tests. At an advance ratio of 0.14 overall reductions were 7.5dB(A) in the continuous loud banging area (descent rate 183m/min) and 4.5dB(A) in the area of most intense noise (219m/min).

The ogee tip (Figure 49) has been tested at both model and full-scale (Mantay[129]). Not only were significant reductions of impulsive noise obtained in many flight conditions but also performance was improved and control loads were reduced. Accompanying tail rotor noise was also lower in some conditions. One of the most significant findings from the UH1 flight tests was a shift of the blade slap envelope (Figure 50) which freed an important glide-slope range from the phenomenon. For example in a 2° descent blade slap reductions of 15dB were measured.

Figure 51 shows a tip developed for future Westland helicopters which is claimed to improve performance and reduce both forms of blade slap by delaying compressibility (Figure 52) and stall effects and by diffusing tip vortex energy (Lowson and Balmford[130]). At $\mu = 0.9$ advancing blade impulsive noise reduction is stated to be 13.5dB.

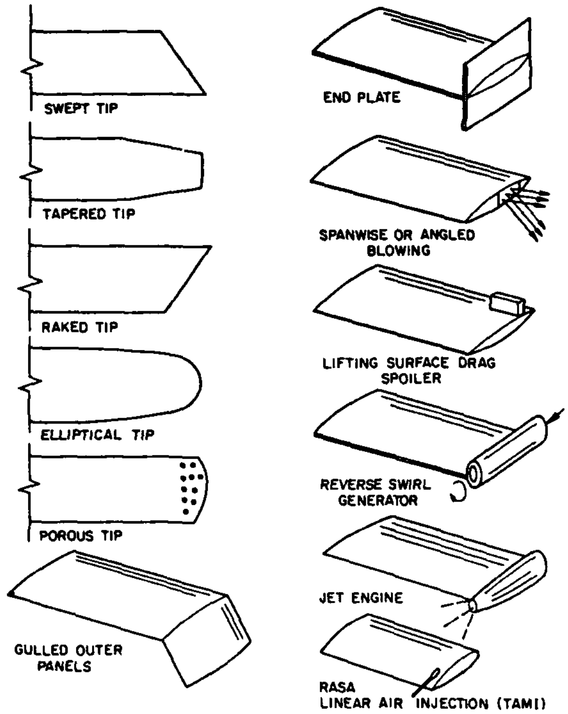


Figure 47 Tip Modifications for Control of Trailing Vortices. From White[128]

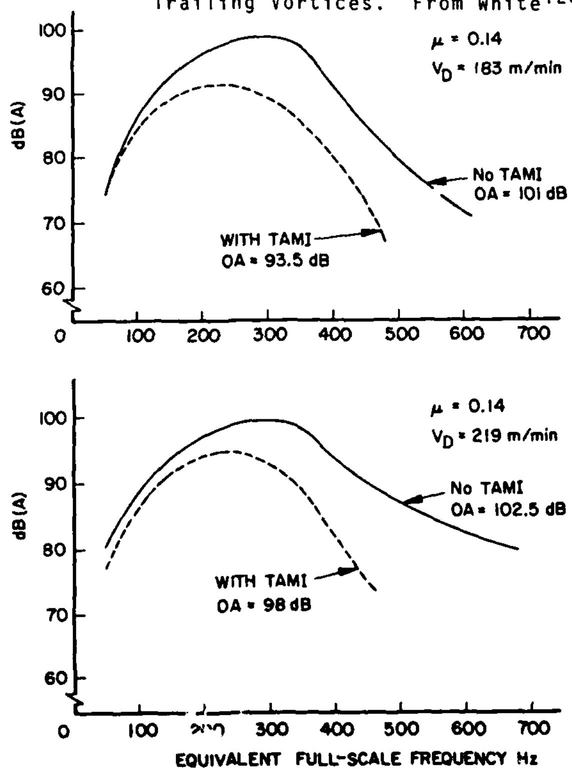


Figure 48 Effect of TAMI on A-weighted Spectrum. From White[128]

Figure 49 Ogee Tip for Tip Vortex Diffusion From Mantay[129]

Tail rotors can make a significant contribution to noise in the mid-frequency range because the fundamental blade-passing frequency is much higher than that of the main rotor (Figure 2). A primary factor is the interaction between the tail rotor and the main rotor wake and this seems to be an important target for noise control efforts. Obviously efforts to diffuse main rotor vorticity will also be beneficial to the tail rotor (as noted above) as will improvements to the geometric configuration.

Leverton et al[131] found that a 'burbling' noise generated by the Lynx tail rotor was due to a main rotor trailing vortices being intersected four or five times as this passed through the tail rotor disc on the advancing blade side. By reversing the direction of tail rotor rotation these intersections were switched to the retreating blades resulting in significant reductions in high-speed flyover noise (Figure 53).

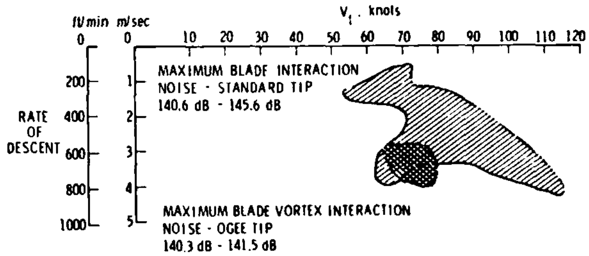


Figure 50 Effect of Ogee Tips on Blade-slap Envelope of UH-1H Helicopter (From Mantay[129])

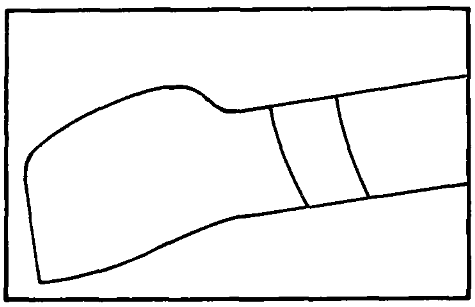


Figure 51 Westland Tip. From Lowson and Balmford[130]

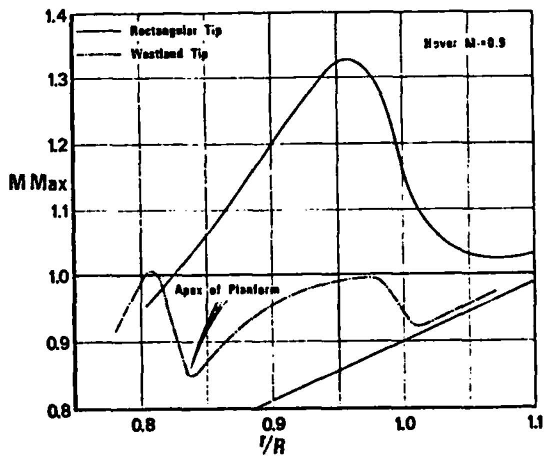


Figure 52 Mach Number Distribution on Westland Tip[130]

After the more obvious noise reduction measures, i.e. minimum tip speed, optimal tip shapes and rotor geometry, etc., the broadband noise could only be reduced further by reducing the 'response' to the residual aerodynamic excitation. A possible technique is to reduce trailing edge noise by the use of porous or variable impedance surfaces (see Hayden and Aravamudan[132] for references) which allows a gradual acceleration of the fluid around the trailing edge (Figure 54). Figure 55 shows noise reductions measured using model airfoils; the dramatic reduction around 150Hz is due to the suppression of vortex shedding noise which is not a practical rotor problem but 5dB reduction of boundary layer/trailing edge interaction noise is also evident.

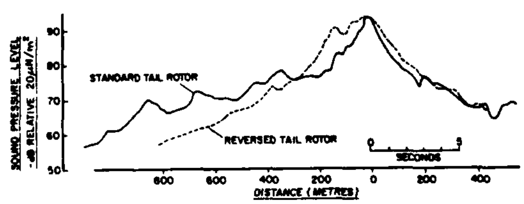
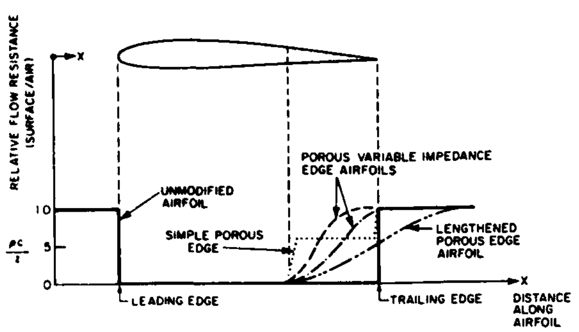


Figure 53 Sound Level Time Histories for Lynx with Standard and Reversed Tail Rotors. From Leverton et al[131]

4.3 Quiet Helicopter Demonstrations

Practical experience in the quietening of helicopters was gained during the U.S. Army's Quiet Helicopter Program reported in the early 1970's[133]. Three rather different helicopters were used, the Hughes OH6A, 954kg (Barlow et al[133]) and the Sikorsky SH3D, 7080kg (Schlegel[134]) both of conventional, main rotor-tail rotor layout and the Kaman HH43B, 4000kg approx., (Bowes[135]) with twin intermeshing main rotors.

Various modifications were made in order to reduce noise from the rotors, engines and transmissions. Intake and exhaust mufflers were applied to the engines which were also better balanced and isolated to reduce vibration levels. Drive systems were manufactured to better tolerances, gear wheels and shafts were damped and gear teeth meshing was imp-



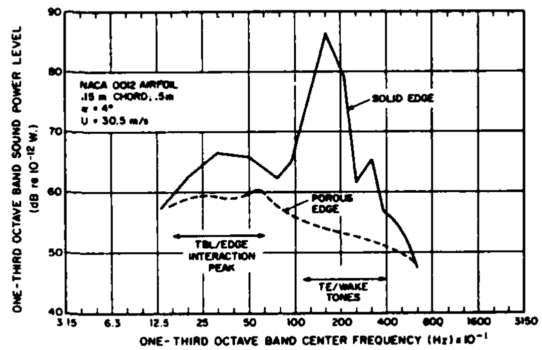
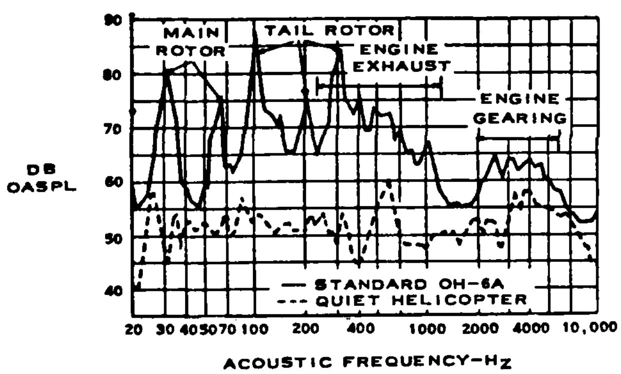


Figure 54 Variable Impedance Trailing Edge. From Hayden and Aravamudan[132]

Figure 55 Effect of Porous Trailing Edge on Broadband Noise of Fixed Airfoil. From Reference 132

roved. Rotor noise levels were reduced by reducing tip speeds, while thrust was maintained by increasing diameters, blade areas, airfoil sections and thrust and sometimes numbers of blades.

Varying degrees of success were achieved with overall level reductions of 12, 8 and 3dB being obtained for the OH6, HH-43B and SH3-D. Figure 56 compares the spectra of the standard and quiet OH6A and Figure 57 shows a mean difference of 8dB between the three



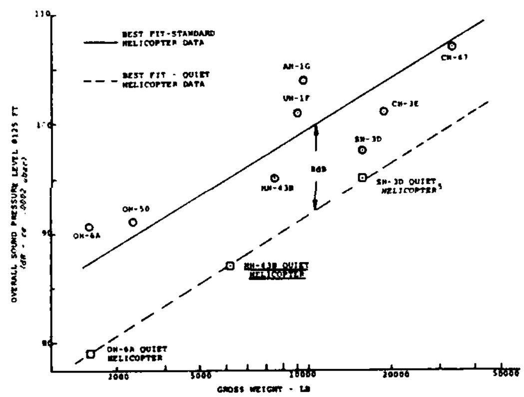


Figure 56 Comparison of Hover Noise Spectra of Standard and Quiet OH-6A

Figure 57 Overall Results of Quiet Helicopter Program; from Bowes[135]

silenced machines and a sample of standard helicopters.

These results are quite impressive if no account is taken of associated performance and cost penalties.

4.4 Costs of Noise Control

In the first published study of the economics of helicopter noise control, Bowes[136,137] estimated the costs and benefits of potential engine and rotor noise reduction on three current types (S61, B205, H500). Although it was concluded that rotor modifications would be excessively costly, 'small but meaningful' cost-effective reductions of turbine exhaust noise could be achieved.

This study was subsequently criticised by the HAA (Wagner[116]) which generated different conclusions summarised in Figure 58. This compares HAA calculations (using Sikorsky methodology) for the S61 with those presented by Bowes. Wagner also stated that a 10dB noise reduction in the OH6 helicopter caused a 46% fall in payload and a 12PNdB reduction in the Boeing 347 (derived from the CH47) a 35% fall.

More detailed cost-benefit analyses have been prepared by Spencer and Sternfeld[127] of Boeing-Vertol who used their in-house noise prediction method to assess noise reduction

treatments for the BO105 (2300kg), Boeing model 179 (YUH61A) (7900kg), CH47 (18500kg), and Model 301 (Heavy lift) (53600kg) helicopters. Detailed noise measurements from these aircraft were analysed to identify the contributions from different sources and to check the adequacy of the prediction model. Various levels of noise control treatment and modification were then defined for each machine, and for each of which noise reductions (in cruise - take-off and approach were not considered), costs (manufacturing and operating), and performance penalties were estimated.

The modifications included (in various degrees) lower rotor speeds, increased blade areas, improved airfoils, more blades, new or modified gearboxes, engine changes, tail rotor repositioning, various structural modifications required to accommodate dynamic/aerodynamic changes, fuselage stretch (tandem rotor machines), as well as many detail changes.

Figure 59 shows, for two helicopters,

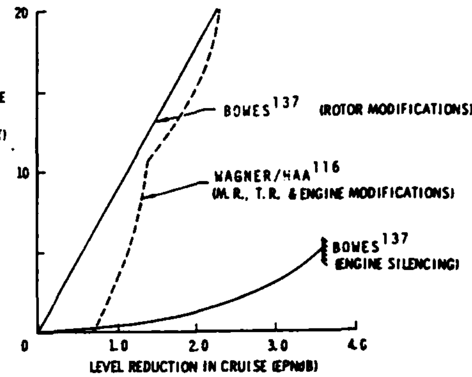
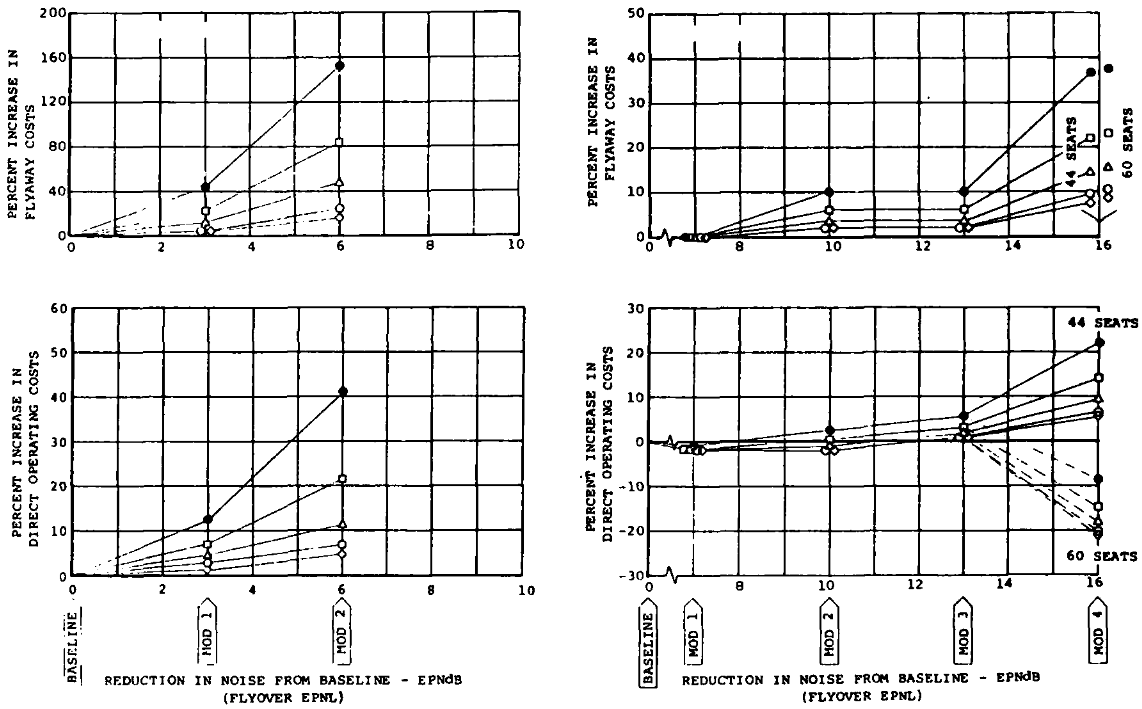


Figure 58 Comparison of Noise Control Cost Estimates from References 116 and 137



(a) Bo 105 (b) CH-47

Figure 59 Effect of Configuration Changes on Flyaway Costs and Direct Operating Costs - of Two Helicopters. (From Spencer and Sternfeld[127])

the increase in both flyaway costs and direct operating costs associated with various degrees of noise control as a function of EPNL reduction and production quantity. The main conclusions of the study were:

1. There were no important general trends, each helicopter type has to be considered as a special case.
2. Modifications to existing production models are more costly than initial design improvements.
3. Production quantity is an extremely important factor and has a greater influence on current machines than new designs.

4.5 Operating Procedures

Noise generation by helicopters, perhaps more so than by fixed-wing aircraft, is very sensitive to flight configuration; significant noise changes occur with weight, speed, turn rates, climb and descent rates and although these effects cannot be generalised, the noise characteristics of particular machines can be systematically logged so

that operators can use them to advantage. Helicopter Association International (HAI) by agreement with the FAA (see below), has published a "Fly Heighbourly Guide"[138] which explains to pilots, operators and heliport managers how to implement helicopter noise abatement practices. (Manufacturers support this initiative by publishing noise abatement procedures in specific rotorcraft flight manuals.) An example shown in Figure 60 is a "fried-egg" diagram which provide guidance on how to avoid blade slap during turns and final approach. Specific procedures do of course have to be defined for individual helicopters and individual heliports.

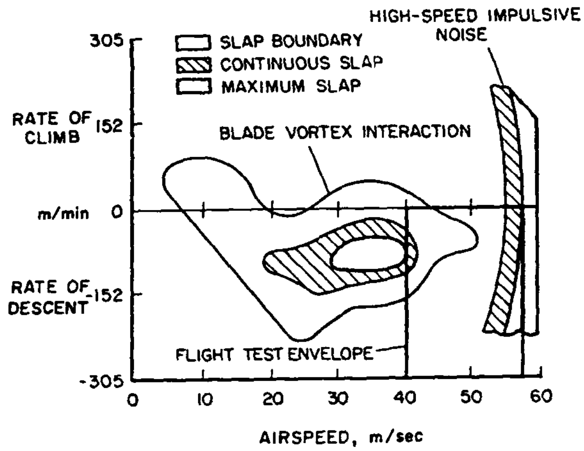


Figure 60 Blade Slap Boundaries for UH-1 Series Helicopters

4.6 Future Prospects

Attempts by aircraft noise certification authorities, particularly the US FAA, to accelerate progress in rotorcraft noise control by developing noise-certification rules was noted in Section 2. The U.S. proposals would have placed stringent limits on current production models, derivatives thereof and new designs. In a comprehensive co-ordinated response[20] to the NPRM the major helicopter manufacturers of the world strongly opposed the proposed rule on the grounds that (a) the limits could not be met economically with available technology* and (b) that the state of knowledge of rotorcraft noise is inadequate to ensure that future designs could meet the noise standards. A plea was made to postpone the legislation, to initiate programs to redefine more reasonable and practicable noise limits, and to conduct extensive research to advance noise reduction technology.

The U.S. legislation was shelved in 1981 on the understanding that the necessary work would be undertaken and that in the meantime, the industry would make serious attempts to control the impact of helicopter noise. In the U.S.A. a co-ordinated government/industry research programme was initiated in 1983 (Raney and Hoad[139]) and HAI launched its voluntary "Fly Neighbourly Program" in 1982[138].

Meanwhile ICAO has been investigating the practical problems of aircraft noise certification. Standardised noise measurements of the Augusta A109 helicopter made in three countries differed by as much as 5EPNdB and the ICAO Noise Committee has organised a 'Round Robin' in which certification measurements are currently being made of the Bell Jet Ranger in eight countries to establish practical repeatability bounds.

5. CONCLUSIONS

The fundamental mechanisms of rotor noise generation now appear to have been identified although theories which will accurately predict the details of the various components are not yet within reach. To date, success has been restricted to the development of relatively crude semi-empirical models whose applicability is perhaps restricted to the modest extrapolation of existing flight data. For example, such methodology has been useful in cost-benefit trade-off studies of rotorcraft noise control.

The current prediction models require as input, spectral distributions of fluctuating blade airloads. The development of analytical procedures by which these could be predicted for design purposes, ab initio, up to the frequencies of acoustic importance seems a remote possibility. The same is true of the quadrupole source fields which assume importance at transonic rotor speeds.

Thus further progress in rotor noise technology would seem best served by more experimentation. Much progress has been made in recent years in the acquisition of flight data, in acoustic wind tunnel experimentation and in acoustic instrumentation and analysis. The necessary acoustic theory, mainly stemming from the basic work of Ffowcs-Williams and Hawkings, is well developed and the major weakness appears to lie in the definition of the source terms.

In the meantime it is perhaps not too optimistic to assume that the available analytical tools will allow reasonably reliable diagnosis of the noise sources on an existing machine. Once these have been quantified, methods could undoubtedly be devised to reduce their strength by design changes to, for example, rotor tip speeds and solidity, airfoil sections and construction, blade tips, rotor pylon location. However, the design of entirely new rotorcraft to meet stringent noise standards seems likely to remain a hazard-

*They pointed out that the proposed limits lie close to the noise levels of existing helicopters (see Figure 11) many of which already incorporate state-of-the-art noise control features including moderate tip speeds and disc loadings, new and thinner airfoils and tailored tip planforms involving sweep and taper. Furthermore, future machines will cruise at higher speeds.

ous occupation for some time to come.

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RECENT DEVELOPMENTS IN THE DYNAMICS OF ADVANCED ROTOR SYSTEMS

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SUMMARY

The problems that have been encountered in the dynamics of advanced rotor systems are described. The methods for analyzing these problems are discussed, as are past solutions of the problems. To begin, the basic dynamic problems of rotors are discussed: aeroelastic stability, rotor and airframe loads, and aircraft vibration. Next, advanced topics that are the subject of current research are described: vibration control, dynamic inflow, finite element analyses, and composite materials. Finally, the dynamics of various rotorcraft configurations are considered: hingeless rotors, bearingless rotors, rotors with circulation control, coupled rotor/engine dynamics, articulated rotors, and tilting proprotor aircraft.

NOMENCLATURE

C_T	rotor thrust coefficient
EI	blade bending stiffness
f	ratio of blade torsional stiffness to pitch link stiffness
K_{P_β}	pitch/flap coupling; positive for flap up, pitch down
K_{P_ζ}	pitch/lag coupling; positive for lag back, pitch down
K_β	flap hinge spring
K_ζ	lag hinge spring
K_θ	pitch hinge spring
r	rotor blade radial station
R	structural flap/lag coupling ($R = 0$ for no coupling, $R = 1$ for complete coupling); or blade radius
V	helicopter forward velocity
β	blade flap degree of freedom
β_d	blade droop angle
β_p	hub precone angle
ζ	blade lag degree of freedom
ζ_s	blade sweep angle
θ	blade pitch angle
θ_f	flexbeam pitch angle
θ_h	hub flap/lag flexure pitch angle
μ	advance ratio (forward velocity divided by rotor tip speed)
ν_β	rotating natural flap frequency, per rev
ν_ζ	rotating natural lag frequency, per rev
σ	damping (real part of eigenvalue, negative for stability); or rotor solidity (ratio of blade area to disk area)
Ω	rotor rotational speed
ω_ϕ	blade pitch frequency, per rev

1. INTRODUCTION

Good dynamic characteristics are a prerequisite for the success of any rotorcraft. Without an adequate aeroelastic stability margin, acceptable rotor and airframe loads, and low aircraft vibration, the machine cannot fulfill its mission, regardless of its performance capabilities. Indeed, the dynamic characteristics often define the operating limits of a rotorcraft. Perhaps more than any other discipline involved in helicopter engineering, the dynamics are very configuration-dependent. In this lecture, the dynamics problems that have been encountered in the development of advanced rotor systems are discussed. The methods of analyzing these problems, as well as past solutions, are reviewed. First, the basic dynamic problems of rotors are discussed; next, advanced topics that are the subject of current research; and finally, the dynamics of various rotorcraft configurations.

In the recent surveys of rotary wing dynamics (Refs. 1-6), the emphasis was on the theory of hingeless rotor stability. Loewy (Ref. 1) provided a discussion of dynamics problems in general. Hohenemser's (Ref. 2) subject was flight dynamics, but he included stability phenomena that involved the fundamental blade modes (hence not vibration); he covered experimental results as well. Friedmann (Ref. 3) gave a good chronological discussion. Friedmann (Ref. 4) dealt with the aerodynamics analysis in particular. Ormiston (Ref. 5) covered bearingless as well as hingeless rotors, and experimental as well as theoretical results. Lowey (Ref. 6) discussed helicopter vibration.

2. BASIC DYNAMIC PROBLEMS

To begin, the basic dynamic problems of rotors will be discussed: aeroelastic stability, rotor and airframe loads, and aircraft vibration. The emphasis will be on describing the primary characteristics of these problems, and the general capability to analyze them.

2.1 Stability

A summary of the basic results from rotary wing stability analysis is appropriate before considering recent developments. Johnson (Ref. 7) provides a complete derivation of these results, as well as references to the original literature.

2.1.1 Flap-Lag Stability

Rotor blade flap-lag stability has received much attention because the lag mode damping is low without a mechanical damper, and because the couplings between flap and lag motion are often complicated in new configurations. With no pitch/lag coupling and a flap frequency of 1/rev, the aerodynamic and Coriolis flap moments due to lag velocity nearly cancel, so the flap equation is decoupled from the lag motion. It follows that the flap-lag motion is stable for an articulated rotor: a rotor with small lag frequency, flap frequency near 1/rev, small pitch/lag and pitch/flap couplings, in hover or low advance ratio. Moreover, the articulated rotor will have a mechanical lag damper (for ground resonance stability). This same canceling of the flap moments due to lag velocity occurs if the flap frequency is above 1/rev with ideal precone, for then the flap spring is not contributing to the coning moment and the coning angle is the same as for a flap frequency of exactly 1/rev. The aerodynamic and Coriolis forces are generally proportional to the rotor thrust, so the lag moments due to flapping are small at low collective. It follows that any flap-lag instability will be a high collective phenomenon.

For a hingeless rotor with no pitch/flap or pitch/lag coupling, or flap/lag structural coupling, it is found that the critical condition for flap-lag stability is zero precone and flap frequency = lag frequency = 1.15/rev. Such a rotor blade is stable with ideal precone and for a flap frequency less than 1/rev or greater than 1.41/rev. The effect of pitch/flap coupling is primarily to introduce the effective flap frequency, including the aerodynamic spring due to pitch/flap coupling, in place of the structural/centrifugal flap frequency in this analysis (note that a flap frequency less than 1/rev then is possible, with negative pitch/flap coupling). The lag mode structural or mechanical damping required for stability increases with the rotor thrust. However, the blade viscous drag damping alone is sufficient to provide stability up to roughly a $C_T/\sigma = 0.10$ for the critical condition defined above. So, in general, any reasonable level of structural damping is sufficient to stabilize a hingeless rotor. In forward flight (at advance ratios of around 0.4), an instability is possible even for an articulated rotor, but, again, the instability is mild, and a moderate amount of lag damping is still sufficient to stabilize the motion.

A flap-lag instability is also possible at high collective pitch due to stall. The loss of flap damping because of the reduced lift-curve slope allows the instability. This phenomenon can only occur in hover, since in forward flight stall is encountered only on part of the rotor disk.

2.1.2 Pitch/Lag, Pitch/Flap, and Flap/Lag Coupling

Flap-lag stability becomes a problem largely because of pitch/lag coupling. For articulated or soft-inplane hingeless rotors (lag frequency less than 1/rev), positive pitch/lag coupling (lag back, pitch down) is destabilizing. For a stiff-inplane hingeless rotor (lag frequency above 1/rev), positive pitch/lag coupling is destabilizing with full flap/lag structural coupling, while negative pitch/lag coupling is destabilizing with no flap/lag coupling (pure inplane and pure out-of-plane modes). With an articulated rotor, the pitch/flap and pitch/lag couplings are determined by the geometry of the root hinges and control system. With a hingeless rotor, in addition to such kinematic couplings, there are effective couplings due to the nonlinear bending and torsion loads on the blade. Structural coupling of the flap and lag motion is produced by pitch of the structural principal axes of the rotor blade. Even a small amount of flap motion in the lag mode as a result of such coupling is very stabilizing because of the high aerodynamic damping associated with out-of-plane motion.

Elementary, but useful, expressions for the effective pitch/lag and pitch/flap couplings can be obtained considering a flap-lag-torsion spring model of a hingeless rotor. A complete derivation is given in Ref. 7. Elastic flap deflection introduces a component of the trim lag moment about the pitch spring; elastic lag deflection introduces a component of the trim flap moment about the pitch spring. For pitch springs inboard and outboard of the droop (representing control system stiffness and blade elastic torsion, respectively), the total nose-up moments are:

3.4 Composites

The use of composite materials for rotor blades and hubs and in helicopter airframes is increasingly common. Composite materials are replacing metals because of better fatigue and strength characteristics for a given stiffness, and better damage tolerance and failure characteristics. For a given bending stiffness (EI), cross section size (z_{max}), and load (M), the maximum stress in a beam is proportional to the modulus: $\sigma_{max} = Mz_{max}/I = (Mz_{max}/EI)E$. Hence, for rotor blade materials a high strength to modulus ratio is desired. Composite materials such as fiberglass and graphite have σ/E values several times that of steel or aluminum. (The lower modulus (E) of composites implies a larger area moment I for a given stiffness. For cases where a large I is unacceptable, titanium may be a good compromise, with σ/E and E values between composites and steel.) Composites can offer advantages in fatigue characteristics and failure modes as well. The use of composite materials also allows additional design flexibility, bringing the opportunity to tailor detailed structural properties as well as overall structural couplings of rotor blades. This additional flexibility will not be fully utilized until the rotor analyses can handle all the unique characteristics of composite materials.

Composite materials are generally orthotropic: the material properties are symmetric with respect to three planes, when one of the coordinates is aligned with the fiber direction. For an orthotropic material, Hooke's law relating the stress tensor to the strain tensor takes the form:

$$\begin{pmatrix} \sigma_{11} \\ \sigma_{22} \\ \sigma_{33} \\ \sigma_{12} \\ \sigma_{13} \\ \sigma_{23} \end{pmatrix} = \left[\begin{array}{c|c} \text{sym-metric} & \text{zero} \\ \hline \text{zero} & \text{diagonal} \end{array}\right] \begin{pmatrix} \epsilon_{11} \\ \epsilon_{22} \\ \epsilon_{33} \\ \epsilon_{12} \\ \epsilon_{13} \\ \epsilon_{23} \end{pmatrix}$$

(For a metal, the material is isotropic: the properties are additionally the same in all three directions, and the matrix above is defined by only three independent parameters.) Now if the fiber direction is at an angle relative to the coordinate system, the constitutive equation takes the form of a monoclinic material. In a monoclinic material, the elastic properties are symmetric with respect to only one plane (for example, the 1-2 plane), with Hooke's law of the form:

$$\begin{pmatrix} \sigma_{11} \\ \sigma_{22} \\ \sigma_{33} \\ \sigma_{12} \\ \sigma_{13} \\ \sigma_{23} \end{pmatrix} = \left[\begin{array}{c|c} \text{symmetric} & \text{zero} \\ \hline \text{zero} & \text{sym-metric} \end{array}\right] \begin{pmatrix} \epsilon_{11} \\ \epsilon_{22} \\ \epsilon_{33} \\ \epsilon_{12} \\ \epsilon_{13} \\ \epsilon_{23} \end{pmatrix}$$

So, in a monoclinic material, the shear stress and strain corresponding to the plane of symmetry (σ_{12} and ϵ_{12} above) are coupled to the normal stresses and strains. A simple coordinate transformation does not change the material from orthotropic to monoclinic. Often, however, a composite structure consists of laminates with different fiber orientations, so the complete structure effectively has only one plane of symmetry. Moreover, the stress in beams (such as rotor blades) is dominated by the components acting on a plane normal to the beam axis (σ_{xx}, σ_{xy}, and σ_{xz} if x is the spanwise variable). So it is the stress and strain relative to the particular coordinate system aligned with the beam axis that is most important. Consequently, it is appropriate to analyze composite beams by assuming monoclinic structural behavior.

Mansfield and Sobey (Ref. 95) developed the bending and torsion stiffness equations for a beam consisting of a fiber composite tube. They examined the structural twist induced by bending and tension.

Worndle (Ref. 96) developed an analysis for the section properties of a composite blade (such as stiffnesses and shear center). The fiber-reinforced material was assumed to be orthotropic, with one axis aligned with the blade span axis, but the other two axes not corresponding to the section axis system. Hence, beam theory was used for a monoclinic material, with the plane of symmetry normal to the span axis. A finite element method was used to solve for the required warping function.

Hong and Chopra (Ref. 97) calculated the flap-lag-torsion stability of a composite rotor blade in hover. The finite element methods of Ref. 88 were used. The equations for a rotating beam were derived following Ref. 20. The composite material introduced the possibility of stiffness terms coupling bending/torsion and tension/torsion. The rotor spar was represented by a box beam, each side consisting of several laminates of composite material, each laminate at a specified ply angle. The section was monoclinic. For horizontal laminates the plane of symmetry was horizontal (the x-y plane; with the x-coordinate

where ϕ is the elastic torsion degree of freedom, and x is the blade radial coordinate. The variable $\hat{\phi}$, rather than the torsion degree of freedom ϕ, was used in the finite element model because the inter-element coupling is defined in terms of the geometric displacement and rotation at nodes. The analysis was restricted to single load path designs. It followed that the centrifugal force was known a priori along each element; the equation of axial force equilibrium could be directly integrated for u; and so the axial displacement u could be eliminated from the equations. Hence the degrees of freedom for each element were: the inplane and vertical displacement and slope (v, v', w, w'), and the geometric twist displacement ($\hat{\phi}$), at each end of the beam (Fig. 87). The displacement within the beam was interpolated using Hermite polynomials: third order for v and w deflections (corresponding to a linear variation of bending moment), and first order for $\hat{\phi}$ (corresponding to a constant torsion moment). Examples were executed for hover, using 10 elements to represent the blade. The trim problem required solution of a nonlinear algebraic equation, with a banded spring matrix. The equations were linearized about the trim solution to calculate stability. The damping matrix was not banded even for a single load path, because of the Coriolis forces (entering an element equation through the axial displacement). The linearized equations were first solved for the free vibration modes, and then these modes were used as degrees of freedom in the stability analysis.

Silvaneri and Chopra (Ref. 44) extended the analysis of Ref. 88 to the case of a rotor blade with multiple load paths at the root--a bearingless rotor. With multiple load paths, the distribution of the centrifugal force among them was not known a priori. So it was no longer possible to eliminate the axial displacement variable and equations from the analysis; moreover, it became necessary to iterate on the calculation of the centrifugal force in the solution procedure. Here the degrees of freedom for each element were: axial displacement (u), inplane and vertical displacement and slope (v, v', w, w'), and the geometric twist displacement ($\hat{\phi}$) at each end; axial displacement (u) at two interior nodes, equally spaced along the element length; and the geometric twist displacement ($\hat{\phi}$) at the element midpoint (Fig. 87). The displacement within the beam was interpolated using Hermite polynomials: third order for v and w deflections and second order for $\hat{\phi}$ (corresponding to a linear variation of bending and torsion moments), and third order for the axial displacement u (corresponding to a quadratic variation of the tension force). The interior node for $\hat{\phi}$ was introduced so the torsion moment variation would be linear, consistent with the bending moment variation. The two interior nodes for u were introduced so the tension variation would be quadratic, consistent with the centrifugal force for uniform mass distribution. The multiple beams at the root and the single beam of the outboard blade were joined at a rigid clevis. The solution procedure followed that of Ref. 88. The mass and stiffness matrices were not banded with a multiple load path at the root. Examples were executed using six to ten elements to represent a bearingless rotor (Figs. 45 and 46).

Rutkowski (Ref. 89) developed a finite element analysis for the rotor flap bending and airframe vertical motion. The equations for the coupled rotor and fuselage motion were linear in this case; hence the trim solution did not influence the stability. The stability was calculated using the free vibration modes (for the entire system, rotor and body), which were obtained from the finite element analysis. A single load path was assumed for the rotor blades; the element equations were therefore similar to those of Ref. 88.

Lefrancq and Masure (Ref. 90) developed a finite element analysis that was used to calculate the frequency and damping of the Triflex tail rotor. The analysis was used to examine the influence of the weight and stiffness of the sleeve.

Borri, Lanz, and Mantegazza (Ref. 91) and Borri, Lanz, Mantegazza, Orlandi, and Russo (Ref. 92) developed a rotor analysis using the finite element representation for azimuthal variations as well as spacial displacements (STAHR, for Stability and Trim Analysis of Helicopter Rotor). The analysis used a finite element or component representation of an isolated rotor blade, including the control system and blade root geometry. The blade response was calculated, and the method was intended to calculate stability as well. By using a finite element representation for the time variation, the dynamics problem was reduced to a nonlinear static problem, to which the conventional techniques of static finite element structural analysis could be applied (for assembly, solution, and data management). It was noted that the computation time became very large when many space-time elements were used.

Giavotto, Borri, Russo, and Ceriotti (Ref. 93) continued the development of the analysis of Refs. 91 and 92. The dynamics problem was formulated as nonlinear algebraic equations. The trim solution was obtained from the constraint that the motion be the same at the beginning and at the end of one rotor revolution. Perturbations of all variables at the beginning and end of one revolution then gave the state transition matrix for a Floquet stability analysis. This state transition matrix was identical to the converged derivative matrix that was already required in the Newton-Raphson solution for the trim response.

Hodges (Ref. 94) developed a finite element computer code (GRASP, for General Rotorcraft Aeroelastic Stability Program) for the stability of a coupled rotor body in hover or vertical flight. Beam and rigid body elements were considered. No small angle assumptions were made in defining the orientations of the elements, and multiple load-path structures could be analyzed. The variable order, nonlinear, slender beam element was assumed to undergo small strains, but the rotations due to deformation could be large. The nonlinear algebraic equations were solved for the static deflection, and then the linearized equations were assembled and solved for the stability.

In applications of the dynamic inflow model, the model has been found to be important more often than not, which is a reflection of the importance of unsteady aerodynamics to rotor aeroelastic problems. Dynamic inflow is a practical way to add unsteady aerodynamics to rotor stability calculations, because it is a finite state model (ordinary differential equations). The simplicity of the model, as well as its fundamental limitations, follow from the assumptions of low frequency and an actuator disk representation.

Peters and Gaonkar (Ref. 86) calculated the influence of dynamic inflow (using the model of Ref. 82) on the flap-lag stability in forward flight. The rigid blade with springs model of a hingeless rotor was used, with no pitch/lag or flap/lag coupling. The influence of unsteady aerodynamics was significant, particularly for the regressing lag mode (Fig. 82).

Johnson (Ref. 80) calculated the influence of dynamic inflow on hingeless rotor ground resonance, comparing with the experimental data of Ref. 47 (discussed above). It was shown that the unsteady aerodynamics were essential for a correct calculation of the body mode damping (Fig. 83). For the matched stiffness configuration, the test identified a low frequency mode that did not correspond to any expected structural mode of the system (Fig. 84; the body pitch and roll modes and the regressing lag mode were accounted for, and the regressing flap mode was known to have too much damping to be observed in this experiment). The calculations associated this mode with the dynamic inflow variables (of course highly coupled with the flap and body motion; Fig. 85), and also predicted that the mode should be measurable for the matched stiffness configuration but not for the other configuration tested (which had a reduced flap flexure stiffness, hence reduced hub moment capability). Thus the existence of this mode in the experimental data, and its successful prediction, confirmed the global, low-frequency character of the unsteady aerodynamics of the rotor--confirmed (for this problem) the fundamental assumptions of the dynamic inflow model.

Warmbrodt and Peterson (Ref. 30) measured the lag damping of a BO-105 rotor in hover. Calculations of the blade stability (using the analysis of Ref. 29) showed the necessity for the dynamic inflow model in this case (Fig. 86). They also noted that it should be possible to directly measure the unsteady aerodynamic influence, since it is predicted to be present in the regressive (low frequency) lag mode but absent in the progressive (high frequency) lag mode.

3.3 Finite Elements

The use of finite element techniques for structural dynamic modeling promises to bring long-needed flexibility to rotorcraft analyses. Such flexibility is particularly desired in the modeling of the rotor hub and blade root, where most of the design innovation appears in new configurations. The survey of Friedmann (Ref. 4) included a review of finite element methods. The use of a finite element method for the free vibration modes of a rotor blade is not uncommon. What still requires development is a full application of finite element discretization to the nonlinear, coupled aerodynamics/dynamics/structural problem of calculating rotor response. A major limitation of rotor analyses based on finite element methods is the very long computing times required. In most applications of finite elements to rotor problems it has been necessary to introduce free vibration modes at some stage, in order to reduce the number of degrees of freedom to a manageable level.

Yasue (Ref. 87) developed a finite element analysis for the flap-lag-torsion response of a rotor to vertical gusts in forward flight. The degrees of freedom for each element were: displacement and slope of the inplane, vertical, and torsion deflection, at each end of the element. The deflection was represented then by third order polynomials within the element. Rigid pitch motion of the blade was introduced at the blade root. The trim solution was obtained from the linear equations for the bending motion only, using a Galerkin technique (not finite elements). Then the gust response was calculated from linearized equations for the free vibration modes, which were obtained from the finite element representation.

Friedmann and Straub (Ref. 25) developed a finite element formulation for the elastic flap-lag stability of a rotor in hover. The spatial dependence of the linearized partial differential equations of motion was discretized using a local Galerkin method of weighted residuals. The degrees-of-freedom for an element were the displacement and slope of the vertical and inplane deflections, at each end of the element (Fig. 87, without torsion). Cubic interpolation polynomials were used for the displacement within an element. Free vibration modes were calculated from the finite element model, and then the nonlinear finite element equations were transformed to modal equations. Finally the nonlinear modal equations were solved for the static equilibrium deflection, and the linearized equations were solved for the stability. The solution was converged with four or five elements used to model a uniform blade; six to eight elements were required when torsion motion was included.

Sivaneri and Chopra (Ref. 88) developed a finite element analysis for the flap-lag-torsion dynamics of a hovering rotor. They obtained the steady state deflection by solving the finite element equations directly (in previous work, the finite element method was used to calculate the free vibration modes, which were then used to solve for the trim blade deflection). The beam equations of Ref. 20 were used. The blade motion was represented by first radial, then inplane, then vertical, and finally pitch deflection (u, v, w, and θ). The pitch θ was comprised of the collective and built-in twist contributions, plus the geometric torsion deflection $\hat{\phi}$:

$$\hat{\phi} = \phi - \int_0^x v''w' \, dx$$

increased at low speed (Fig. 79). The influence of the regulator on blade loads was similar to that in Ref. 76.

Wood, Powers, Cline, and Hammond (Ref. 78) reported results from flight tests of this regulator on an OH-6A helicopter. Again, the adaptive, open-loop regulator with caution was used. The 4/rev vertical and lateral vibration were reduced significantly (Fig. 80), but the longitudinal vibration increased at high speed. The pitch link loads were increased by the presence of higher harmonic control, and there was some increase in blade bending loads (Fig. 81).

Gupta, Wood, Logan, and Cline (Ref. 79) conducted further higher harmonic control flight tests with the OH-6A. The controller software was changed from fixed point to floating point for better accuracy. The system and measurement noise variances in the Kalman filter were adjusted in flight to optimize the identification. The result of these changes was a significant improvement in the performance of the regulator (Fig. 80). The control system reduced the cockpit vibration on all three axes simultaneously.

3.2 Dynamic Inflow

Despite the long-established importance of unsteady aerodynamic forces in aeroelastic phenomena, an entirely satisfactory model for rotary wing unsteady aerodynamics is still not available. A model for the noncirculatory loads can be readily obtained from two-dimensional unsteady airfoil theory, but the results from either two-dimensional or fixed-wing three-dimensional wing theory for the circulatory loads are not applicable since the wake models are not correct for rotary wings. The development of a general theory for rotor unsteady airloads is difficult because of the complex geometry of the rotor wake, even in hover. Moreover, in forward flight a time-invariant model of the system is not possible because of the periodically varying aerodynamic environment of the blades. Consequently, attention has recently been focused on the development of relatively simple models for the unsteady aerodynamics. These models take the form of differential equations for parameters defining the wake-induced velocity distribution over the rotor disk, hence are referred to as dynamic inflow models. Johnson (Refs. 7 and 80) and Gaonkar and Peters (Ref. 81) provided a derivation and discussion of dynamic inflow.

Typically, the wake induced velocity perturbation is represented by a linear variation over the rotor disk:

$$\delta\lambda = \lambda_u + \lambda_x r \cos\psi + \lambda_y r \sin\psi$$

where r and ψ are the polar coordinates of the rotor disk, so λ_u defines a uniform variation of the induced velocity, while λ_x and λ_y define longitudinal and lateral variations, respectively. The induced velocity variables are related to the net aerodynamic loads on the rotor (thrust C_T, pitch moment C_{M_y}, and roll moment C_{M_x}) by a first order differential equation:

$$\tau\dot{\lambda} + \lambda = (\partial\lambda/\partial L)L$$

where

$$\lambda = (\lambda_u\ \lambda_x\ \lambda_y)^T$$

$$L = (C_T\ -C_{M_y}\ C_{M_x})^T$$

This is a low-order, global model of the rotor unsteady aerodynamics. The model can only represent low-frequency effects. Note that with no flap-hinge offset, an articulated rotor cannot sustain a hub moment. Hence, the linear components of the dynamic inflow model are primarily important for hingeless rotors.

The derivative matrix $\partial\lambda/\partial L$ can be obtained from differential momentum theory (see Peters (Ref. 82) and Johnson (Ref. 7)) or from unsteady actuator disk theory (see Miller (Ref. 83), Pitt and Peters (Ref. 84), and Johnson (Ref. 7)). Rotor response and stability measurements, as well as parameter identification work, generally support the values so obtained for hover. The inflow dynamics model is not as well verified for forward flight as for hover, but the model of Pitt and Peters (Ref. 84) is proving to be very effective (see Ref. 81).

Typically, the time lag is written $\tau = \kappa(\partial\lambda/\partial L)$, where κ is a diagonal, constant matrix. The values for the constants in κ are obtained by considering the apparent mass of an impermeable disk subject to linear or angular motion (see Peters (Ref. 82) and Pitt and Peters (Ref. 84)). These estimates are supported by experimental data and parameter identification, at least to within a factor of two (Ref. 80). Without the time lag ($\tau = 0$), a quasi-static inflow model is obtained, the effects of which are expressed by a lift deficiency function (Refs. 7 and 80). For rotor dynamics problems in which the dominant aerodynamic forces are the lift perturbations due to angle-of-attack changes, the magnitude of the aerodynamic influence is described by the blade Lock number (which contains the section lift curve slope). Hence, in such cases, the effects of the quasi-static inflow model can be largely represented by the use of an effective Lock number that is the product of the actual Lock number and the lift deficiency function (see Curtiss and Shupe (Ref. 85)). A quasi-static inflow dynamics model has long been used in handling qualities analyses. The need for the time lag appears to depend on the specific problem involved. The influence of the time lag is often enough to be measurable, but the quasi-static model may still give qualitatively correct results.

where

$$C = -DT^TW_z$$

$$D = (T^TW_zT + W_\theta + W_{\Delta\theta})^{-1}$$

This is referred to as an open-loop control law in Ref. 75, since it is based on the uncontrolled vibration level z_0. In the adaptive system, z_0 is in fact identified from the current vibration measurement z_n. An alternative form of the control law is:

$$\Delta\theta_n = Cz_{n-1} - DW_\theta\theta_{n-1}$$

which involves the direct feedback of the measured vibration, whether the system parameters are identified or not (referred to as a closed-loop control law; see Ref. 75 for further details).

Limiting the control amplitude by use of the weighting matrix W_θ has the effect of reducing system effectiveness at all conditions. It is preferable that the control authority be sufficient to allow full use of the active alleviation system. Implementation of external constraints on rate of change of control leads to poor dynamic response; such limits should be an integral part of the control law. Limiting the rate of change of the control by use of the weighting matrix $W_{\Delta\theta}$ is very beneficial: the dynamic response of the system is unacceptable without it and good with it, yet a control rate limit does not change the final optimum solution.

The above derivation of the control law assumed that the parameters are known; the result was a deterministic controller. With unknown estimated parameters, the certainty-equivalence principle may be applied: the deterministic control solution is simply used with the estimated paramater values. Alternatively, the possibility of errors in the parameter estimates can be acknowledged by minimizing the expected value of the performance index:

$$J = E(z_n^TW_zz_n) + \theta_n^TW_\theta\theta_n + \Delta\theta_n^TW_{\Delta\theta}\,\Delta\theta_n$$

which produces a cautious controller. The result is equivalent to introducing a weighting matrix on the control amplitude or rate, proportional to the parameter error-variance (which is calculated in the course of the Kalman filter identification procedure; the <u>a priori</u> estimate error, $M_n = P_{n-1} + Q_{n-1}$, is used). With the closed-loop form of the control law, caution introduces an effective limit on the control rate ($W_{\Delta\theta}$). With the open-loop form of the control law, caution introduces an effective limit on the control amplitude (W_θ). Reference 75 provides complete details. Hence the cautious controller provides a means to automatically introduce control limits to compensate for uncertainty in the parameter estimates. That the controller development process need not be concerned with picking the weighting matrix is an advantage; that the designer no longer has the option of selecting the weighting matrix is a disadvantage. Use of a rate limit ($W_{\Delta\theta}$) is probably always desirable, but is not provided by the caution with the open-loop form of the control law. There is an additional problem (that may be of more theoretical than practical concern). The number of parameters to be identified will be much greater than the number of measurements, implying an identifiability problem: there may not be enough information in the measurements to identify all the parameters individually. Simulations show that the feedback control system generally is well behaved, which may be explained by the view that it is necessary to accurately identify the control values required for minimum vibration (about the same in number as the number of measurements), not the individual values of all the parameters. The identifiability problem can, however, be reflected in large values of the parameter error-variance in the Kalman filter, which would introduce large control limits with the cautious controller.

The two options for identification (off-line or adaptive) and the two options for control (feedback of z_0 or z_n; i.e., open loop or closed loop) implies four configurations for the self-tuning regulator (Ref. 75). The configuration of interest here is the adaptive open-loop regulator (Fig. 76), consisting of on-line identification of the parameters and calculation of the gain matrix, with control based on the identified value of the uncontrolled vibration level z_0. This system has been investigated analytically, and in both wind tunnel and flight tests. The configuration tested was the cautious controller, without explicit limits on control amplitude or rate (W_θ or $W_{\Delta\theta}$).

Hammond (Ref. 76) tested a four-bladed, model articulated rotor in a wind tunnel, with the adaptive open-loop regulator. The cautious controller form was required for smooth operation during changing test conditions. Vibration alleviation capability was assessed in terms of the vibratory hub moments and vertical shear (Fig. 77). A 70%-90% reduction of the 4/rev vertical force was achieved, and a 30%-70% reduction of the 4/rev pitch moment; the 4/rev roll moment was reduced only slightly. The higher harmonic control produced some decrease in blade flapwise bending loads; and an increase in edgewise bending, torsion, and pitch link loads (Fig. 78).

Molusis, Hammond, and Cline (Ref. 77) extended the work of Ref. 76 by using feedback of acceleration rather than hub load measurements. The regulator was tested in steady-state operating conditions; with varying wind-tunnel speed; and with collective pitch variations. Generally, the conditions for minimum vibration levels did not correspond to those for minimum oscillatory hub loads; it was essential to sense the vibration directly. The cautious controller showed smooth operation and good tracking ability. The vertical and longitudinal vibrations were reduced significantly, but the lateral acceleration was

The present discussion will concentrate on the concept that has been carried to flight tests--the self-tuning regulator. This approach is based on the use of blade pitch motion at harmonics of the rotor speed above 2/rev in the rotating frame, and hence is referred to as multicyclic or higher-harmonic control. Among the early work was a test reported by McCloud and Kretz (Ref. 73) and Kretz, Aubrun, and Larche (Ref. 74), of multicyclic control on a full-scale jet-flap rotor in a wind tunnel. Johnson (Ref. 75) gives a full derivation and comparison of the various self-tuning regulators for multicyclic control of helicopter vibration. Only the version of the regulator that has been flight tested will be discussed here.

A self-tuning regulator is a control system combining recursive parameter estimation with linear feedback. As developed for multicyclic control of helicopter vibration and loads, this regulator is characterized by (1) a linear, quasi-static, frequency-domain model of the helicopter response to control; (2) identification of the helicopter model by a Kalman filter; and (3) a quadratic performance function controller. Figure 75 outlines the control task: minimize airframe vibration (and perhaps loads or even power) using blade pitch control (defined in the rotating or nonrotating frame).

The requirement for an adaptive control system, in which the parameters describing the helicopter model are identified on line, follows from the inability of current analytical tools to predict vibration characteristics with sufficient accuracy; and from sufficient sensitivity of the vibration characteristics to changes in aircraft configuration and flight condition that a prescribed-parameter control system would be ineffective or at least inefficient.

It is assumed that the helicopter can be represented by a linear, quasi-static frequency-domain model relating the output vector z (consisting of harmonics of the vibration) to the input vector θ (consisting of harmonics of blade pitch control) at time $t_n = n\,\Delta t$:

$$z_n = z_o + T\theta_n$$

The sampling interval Δt must be long enough for transients to die out and the harmonics to be measured. The assumption of linear response to control is supported by the experimental observation that only a small multicyclic control amplitude (of the order of 0.5 to 1.5 deg) is required for vibration alleviation. There is theoretical and experimental evidence both supporting and contradicting the assumption of linearity. There are observed phenomena that may be attributable to nonlinearity, and there are arguments that an iterative linear model is appropriate even for a nonlinear system.

The uncontrolled vibration level z_o and the control matrix T are not known; the state-of-the-art of vibration prediction is not sufficient to allow their calculation to the required accuracy. Hence, an adaptive control system is required, in which these parameters are to be identified during the control process. Grouping the unknown parameters into a single matrix and introducing measurement noise gives:

$$z_n = [T\ z_o](\theta_n^T\ 1)^T + v_n$$

The measurement noise v_n is assumed to have zero mean, variance r_n, and Gaussian probability distribution. The lack of knowledge of the system parameters will be expressed by modeling them as a random process:

$$[T\ z_o]_{n+1} = [T\ z_o]_n + U_n$$

where U_n is a random variable with zero mean, variance Q_n, and Gaussian probability distribution. This equation implies that the parameters vary and that the order of the change in one time-step can be estimated, but no information is available about the specific dynamics governing the variation of the parameters. Then the minimum error-variance estimate of the parameters is obtained from a Kalman filter:

$$[\hat{T}\ \hat{z}_o]_n = [\hat{T}\ \hat{z}_o]_{n-1} + (z_n - \hat{z}_{on-1} - \hat{T}_{n-1}\theta_n)k_n^T$$

The Kalman gains are obtained from:

$$k_n = P_n(\theta_n^T\ 1)^T/r_n$$

$$P_n^{-1} = (P_{n-1} + Q_{n-1})^{-1} + (\theta_n^T\ 1)^T(\theta_n^T\ 1)/r_n$$

where P_n is the variance of the error in the estimate, after the measurement. (See Ref. 75 for further details; the estimation of parameters is in fact done by rows.)

The control algorithm is based on the minimization of a quadratic performance index:

$$J = z_n^T W_z z_n + \theta_n^T W_\theta \theta_n + \Delta\theta_n^T W_{\Delta\theta}\,\Delta\theta_n$$

where $\Delta\theta_n = \theta_n - \theta_{n-1}$. Typically, the weighting matrices are diagonal, so J is a weighted sum of the mean squares of the vibration, loads, and control. The matrices W_θ and $W_{\Delta\theta}$ constrain the amplitude and the rate of change of the control, respectively. The control law is obtained by substituting for z_n, setting the derivative of J with respect to control to zero, and solving for θ_n:

$$\theta_n = Cz_o + DW_{\Delta\theta}\theta_{n-1}$$

rotors (Fig. 59). This theory significantly underpredicted oscillatory pitch link loads above 60 knots, because the measured loads were predominantly 1/rev while the calculated loads were 3/rev. Sheffler (Ref. 60) was able to predict blade bending loads well even at high thrust and high speed (Fig. 60). The prediction of the torsion moment at high thrust was poor even at moderate speed however (Fig. 62), because stall was encountered too early in the theory and thereafter the calculated load did not increase much with thrust.

Yen and Weller (Ref. 51) calculated loads on an articulated rotor, both with standard blades and with tabbed blades (Fig. 57). Their results illustrate well the limits of current load prediction capability. The calculated mean and oscillatory pitch link loads agreed well with test (Fig. 63). Yet examination of the corresponding time histories (Fig. 64) reveals that the fundamental phenomena involved were not being modeled correctly: the tab changed the measured time history dramatically, while the predictions for the two blades had very similar character.

2.4 Vibration

Rotor-induced vibration prediction adds an increased importance of the airframe structural dynamics to the many disciplines already required for loads prediction. Reichert (Ref. 64) discussed various means for vibration control on helicopters: pendulum or bifilar absorbers on the blades or hub; rotor isolation, anti-resonance isolators, or absorbers on the airframe; and higher harmonic control. There has been a clear trend of reduced helicopter vibration in newer designs--after vibration treatment has been included (Fig. 65). Reichert made the point that analysis of vibration is not adequate, and identified the need to model the airframe as well as the rotor.

Cronkhite (Ref. 65) compared NASTRAN predictions and shake test results for AH-1G airframe structural dynamics. A model intended to be valid up to about 30 Hz (above 5/rev) was developed. The structural damping was particularly difficult to quantify for the analysis; here 2% critical damping was assumed. In the frequency response above 20 Hz, the tests showed more damping than predicted (Figs. 66 and 67).

Stoppel and Degener (Ref. 66) compared NASTRAN predictions and shake test results for BO-105 airframe structural dynamics. A large order model was needed for accuracy. A frequency error of about 5% was achieved for the lowest modes, and 10% for the higher modes (Fig. 68). It was noted that the large concentrated masses typical of helicopter structures result in many modes at relatively low frequency (specifically, below 8/rev or about 56 Hz for the BO-105).

Gabel, Reed, Ricks, and Kesack (Ref. 67) developed a NASTRAN finite element model of a CH-47D airframe, emphasizing the prediction of structural vibration. They documented the planning and development of the NASTRAN model, the predictions, the shake test, and the correlation of the measured and predicted response. The correlation below about 20 Hz was considered good by the standards of the industry (Fig. 69), particularly the forced vibration at 3/rev and 6/rev. The predictions at high frequency were not considered good. The calculations were based on 2.5% structural damping for all modes. Correct modeling of the damping would improve the prediction of peak amplitudes (for example, in the lateral response shown in Fig. 69). To improve the correlation of mode shape and frequency, it would be necessary to increase the detail of the structural modeling (including the test fixture), and of the mass distribution. Figure 70 illustrates the achievable improvement.

Sopher, Studwell, Cassarino, and Kottapalli (Ref. 63) compared predicted and measured vibration for a wind tunnel test of an articulated model rotor (representative of the UH-60). Two representations of the rotor support structure were considered: a three-mode model, consisting of the modes nearest 4/rev (between 3.4 and 5.2/rev), and a nine-mode model, consisting of all body modes from 1.9 to 6.5/rev. The trends of the vibration with forward speed and higher harmonic control were predicted satisfactorily, but the absolute levels were not correct (Fig. 71). The predicted absolute levels of vibration were sensitive to the body modal characteristics. The vibration change due to a prescribed higher harmonic control input was underpredicted, although the results were improved using the nine-mode body model (Fig. 72). Comparison was also made between the experimentally determined optimum vibration using higher harmonic control and the theoretically determined optimum (not using the measured values of the control inputs).

Gabel and Reichert (Ref. 68) examined the use of pendulum absorbers to reduce the vibration level on the BO-105 helicopter in transition, between 20 and 40 knots (Figs. 73 and 74). Flap and lag pendulum absorbers for 3/rev vibration were used. A blade tuning weight, to get the third flap mode below 5/rev, was also used; the tuning weight was not effective alone however. The final configuration of pendulum absorbers and tuning weights was 1.25% of the gross weight.

3. ADVANCED TOPICS IN DYNAMICS

Next, advanced topics that are the subject of current research will be considered: vibration control, dynamic inflow, finite element analyses, and composite materials.

3.1 Vibration Control

Passive control of helicopter vibration, such as discussed above, has the disadvantages of significant weight penalty and lack of flexibility. There are numerous concepts for active control of vibration, for example, Ham (Ref. 69), Kretz (Ref. 70), Gupta and DuVal (Ref. 71), and Jacob and Lehmann (Ref. 72).

found that these couplings did increase the damping, but were not sufficient to stabilize the case with a soft flap flexure (Fig. 47). The instability was less severe for the matched stiffness configuration, so pitch/lag coupling was indeed able to provide stability. The theory of Ref. 38 was used to analyze this configuration. Generally, the body mode damping was underpredicted by the theory; it was speculated that this discrepancy was due to unsteady aerodynamics. The theory tended to be less accurate at high collectives. The measured damping increased with collective (except at an air resonance instability), while the theory did not (particularly at the higher rotor speeds).

Yeager, Hamouda, and Mantay (Ref. 48) conducted a wind tunnel test of a model soft-inplane hingeless rotor, with body pitch and roll motion. The measurements showed the favorable influence of precone or negative droop on the stability. Calculations using the method of Ref. 29 gave good correlation with the experimental data, for both hover and forward flight (Figs. 48 and 49).

Hooper (Ref. 49) used the analysis of Ref. 38 to examine the influence of blade-design parameters on the air resonance and ground resonance stability of a bearingless rotor. The configuration considered involved a single-beam flexure, and a rigid torque tube with a single shear restraint (pitch link) on the flexbeam axis (for zero coupling). Air resonance instability was predicted for high rotor speed and high collective, at the resonance with the regressing flap mode. Hub and blade sweep, hub and blade prepitch, and hub precone did not change the stability significantly. Blade negative droop or vertical offset of the torque tube shear restraint above the flexbeam were stabilizing. Both parameter changes produced negative pitch/lag coupling, which stabilized the blade flap-lag motion sufficiently that the air resonance instability occurred only in a narrow rotor speed range at resonance (which occurred at a fairly low rotor speed as well). The ground resonance instabilities, at resonances with the body pitch and roll modes, were more severe and less sensitive to collective than the air resonance instability. Negative droop or vertical offset of the shear restraint stabilized the pitch mode resonance (at low rotor speed) but destabilized the roll mode resonance (at high rotor speed). At low collective even the pitch mode was destabilized slightly by these parameters. It was concluded that it would be necessary to design the helicopter to avoid the roll mode, or use damping augmentation.

2.3 Loads

Regarding the prediction of rotor and airframe structural loads, it is not possible to identify a single assumption, a single limitation, or a single discipline that dominates the problem. For a good prediction of loads it is necessary to do everything right, all of the time. With current technology it is possible to do some of the things right, some of the time. Most of the recent development of rotor dynamics theory has concerned the stability problem. The loads analysis requires the full nonlinear solution, not just the linearized equations, and demands much more attention to the aerodynamics.

Piziali (Ref. 50) made the observation that there had been significant progress in the development of rotor aeroelastic computer simulations, but that the progress had been primarily in expanding the scope of the predictive capability. During the previous decade, the improvement in correlation between measurements and predictions had not been significant. Piziali was speaking at the 1973 AGARD conference on rotor loads, and referring to progress since the initial work with nonuniform inflow calculations in the early 1960s. The statement is equally valid now; recent advances have been in scope, not accuracy. It is possible to make loads predictions for the new rotor concepts that are of sufficient accuracy to support engineering design and development. Yet the level of accuracy for these predictions is about the same as the capability for today's conventional rotors when they were new; and the correlation of measured and predicted loads on conventional rotors has not improved significantly either. Piziali also made the comment that the technology level did not then limit the structural representation, but it did limit the aerodynamic representation; the participants at the conference did not agree. Reichert agreed that the limiting factor in the aerodynamics is obtaining a fundamental model of the phenomena, and he added that the limiting factor in the dynamics is the complexity of the code. Carlson and Kerr emphasized the multidisciplinary nature of the loads problem. Yen and Weller (Ref. 51) remarked that a good prediction of loads requires an accurate representation of the structure, state-of-the-art aerodynamics, plus the user's knowledge and experience with the analysis. So there is still some art as well as science in the task.

The AGARD conference of 1973 (Refs. 52-58) provided a good summary of rotor loads predictive capability. Generally the computer codes used then are still the primary design tools of the industry. Figures 50-55 illustrate the predictive capability for articulated, teetering, and hingeless rotors. More recent efforts (Refs. 51 and 59-62) have produced similar results (Figs. 56-60).

Sopher, Studwell, Cassarino, and Kottapalli (Ref. 63) compared predicted and measured loads for a wind tunnel test of an articulated model rotor (representative of the UH-60). Calculated edgewise bending loads were higher than measured (Fig. 61), due to a high 5/rev predicted load. The predicted edgewise first elastic bending mode was at 5/rev, so there was a significant increase when body motion was included in the analysis. The calculated torsion loads were low, because of underprediction of the 1/rev to 4/rev harmonics. The sensitivity of the loads to prescribed higher harmonic control changes was significantly underpredicted (but the measured edgewise loads showed a significant 6/rev component, so the control in the rotating frame might not have been pure 3/rev).

It is more difficult to predict blade torsion and pitch link loads than to predict bending loads. The torsion forces (aerodynamic, inertial, and structural) are higher order than the forces responsible for bending loads, and blade stall is particularly significant for torsion loads. Staley (Ref. 59) calculated hingeless rotor loads using a computer code originally developed for teetering and articulated

a torque rod). Figure 39 shows the influence of precone and droop (such that the total $\beta_p - \beta_d = 2.5°$) and flexbeam prepitch (with the corresponding flexbeam to blade angle set so that the net blade built-in pitch was 7.95°). The data showed that negative droop (producing negative pitch/lag coupling) was preferred to positive precone. There was a small, but favorable, effect of flexbeam pitch angle on air resonance stability because of the increased structural coupling.

Lytwyn (Ref. 40) developed an analysis of bearingless rotor stability. A modal representation of the blades was used, and Floquet theory was used in forward flight. Figures 40 and 41 compare calculated and measured stability for the bearingless main rotor (BMR) on a whirl stand and in flight. Warmbrodt, McCloud, Sheffler, and Staley (Ref. 41) conducted a full-scale wind-tunnel test of the BMR. Hover and forward flight stability measurements were compared with the predictions of Ref. 40 (Fig. 42).

Dawson (Ref. 42) conducted a hover test of model bearingless rotor inplane mode stability (with no body motion). The experiment was designed to verify the predictions of Ref. 38. The rotor had a single flexbeam with an external torque tube. The configurations tested were mainly at zero precone, with the pitch link arm at the leading edge, the trailing edge, or both. With both leading and trailing edge pitch link arms, the blade was stiffer in torsion and there was no pitch/flap coupling. With only the leading edge or trailing edge arm, one position tested was near the equivalent flap hinge radial station. The influence of pitch/flap coupling was examined by varying the radial location of the pitch arm. The rotor was stiff-inplane at low rotational speed and soft-inplane at high speed. Figures 43 and 44 show the influence of pitch link location and precone/droop; the calculations were obtained from the analysis of Ref. 38 (FLAIR). Figure 43 also shows predictions for one case based on the analysis of Ref. 36 (G400). A pitch-flap flutter instability was encountered in some cases; for example, it occurred in one configuration (trailing edge pitch link, positive pitch/flap coupling, precone) near zero collective and high rotational speed at a resonance of the first torsion and second flap bending mode frequencies. The theory did not predict such instabilities because it lacked higher blade bending modes and unsteady aerodynamics. In general, the experimental results were more complicated than anticipated, with frequent encounters of high blade loads at moderate collective, and pitch-flap flutter.

Bousman and Dawson (Ref. 43) elaborated on the flutter results from hover tests of two- and three-bladed bearingless rotors (Ref. 42). Three types of flutter were identified, all involving little blade lag motion. The first type involved the second flap mode and first torsion mode, and was considered a classical flap-pitch flutter since it occurred around 2.5/rev and at all collective pitch angles. The second type was a single degree-of-freedom flutter of the first torsion mode, occurring above 3/rev and at low collective. The third type was a single degree-of-freedom flutter of the regressing flap mode (for the three-bladed rotor only), occurring slightly above 1/rev and at low collective. Since the purpose of the test was to measure those dynamic characteristics involving the lag motion, a systematic examination of the influence of operating condition and blade parameters on the flutter instabilities was not conducted. Type 1 flutter occurred only with a leading edge pitch link. Type 2 flutter occurred only with an inboard, trailing edge pitch link (positive pitch/flap coupling). Type 3 flutter occurred only with the three-bladed rotor, with an outboard, trailing edge or inboard, leading edge pitch link (negative pitch/flap coupling). The blade configuration with both leading and trailing edge pitch links had a high-torsion frequency, so never encountered flutter. The occurrence of a single degree-of-freedom instability at low collective suggested a wake-excited flutter. The type 3 instability indeed occurred very near 1/rev, and in the correct mode for wake-reinforcement of the unsteady aerodynamic forces. The type 2 flutter, however, was not observed in a mode that would be expected to be associated with wake reinforcement.

Sivaneri and Chopra (Ref. 44) applied a finite element analysis to the calculation of bearingless rotor flap-lag-torsion stability in hover. Significant differences were found between a solution modeling correctly a multiple load-path blade root, and a solution for an equivalent single beam (Figs. 45 and 46). The differences were traced to the nonlinear stiffness elements coupling the flap-lag-torsion motion. In particular, when the pitch of the twin beams at the blade root varied with collective, the flap and lag stiffness and the flap/lag coupling varied. The equivalent properties of the single-beam model could not be defined to reproduce this behavior, hence could not match the the correct blade frequencies at all collective pitch values.

Chopra (Ref. 45) used a finite element analysis to calculate stability for the bearingless rotor model of Ref. 42. The results are shown for one case in Fig. 43. The analysis used seven elements (three on the blade and two each on the flexbeam and the torque tube), plus a spring representing the pitch link.

2.2.6 Ground and Air Resonance

Ormiston (Ref. 46) developed an analysis of ground and air resonance, using the rigid blade with hinge spring model. It was found that pitch/lag and flap/lag coupling did not affect ground resonance at zero thrust, but could stabilize ground resonance at high thrust. These couplings were also predicted to stabilize air resonance, hence could be alternatives to lag-mode structural damping.

Bousman (Ref. 47) conducted a hover test of air resonance stability using a model hingeless rotor (a ground-based test, but the body frequencies were appropriate for air resonance rather than ground resonance). The rotor of Ref. 18 was used, with rigid blades and flap/lag flexures. The rotors were soft-inplane at the resonances with the body pitch and roll modes. Two cases were considered, one matched stiffness and one with the flap flexure much softer than the lag flexure. The test was intended to check the predictions of Ref. 46 that pitch/lag and flap/lag coupling could stabilize air resonance. It was

2.2.4 Forward Flight

Miao and Huber (Ref. 13) conducted a wind-tunnel test of a model soft-inplane hingeless rotor with body pitch and roll freedom. The tests showed a favorable effect of reduced precone in forward flight (Fig. 31), just as in hover. The rotor stability increased with forward speed (Fig. 32).

Peters (Ref. 31) analyzed the influence of forward flight on flap-lag stability, using the rigid blade with hinge spring model of Ref. 8. The effects of precone, pitch/flap coupling, and pitch/lag coupling were generally the same at moderate advance ratio as at hover. Figures 33-35 show the influence of advance ratio on flap-lag stability (see Figs. 3-5 for hover). For a particular combination of flap and lag frequencies, the stability was significantly changed above about $\mu = 0.4$. The influence of the periodic coefficients on the stability was not great, however, until much higher speeds (Fig. 36).

Friedmann and Silverthorn (Ref. 32) analyzed flap-lag stability of an elastic blade at high advance ratio. They included structural flap/lag coupling and reverse flow, and linearized the equations about the equilibrium position in hover. Friedmann and Shamie (Ref. 33) continued the analysis of elastic flap-lag stability in forward flight. They linearized the equations about the time-varying blade response of forward flight, which produced much different results than the linearization about hover (Ref. 32). The rotor was trimmed to specified thrust and propulsive force, and zero hub moment (propulsive trim); or trimmed to zero hub moment for fixed collective and shaft angle (moment trim). The stability solution was somewhat sensitive to the trim method, particularly for soft-inplane rotors. Only the blade flap motion was considered in the trim solution.

Friedmann and Kottapalli (Ref. 34) analyzed hingeless rotor flap-lag-torsion stability in forward flight. Reverse flow was included in the aerodynamics, but not stall. Forward flight introduced more variables in the rotor trim solution, and the trim blade deflections were time varying (periodic functions of the rotor azimuth). Moreover, the linearized equations for stability calculations had periodic coefficients. They trimmed the rotor thrust and propulsive force, with zero pitch and roll moments about the helicopter center of gravity, using only the blade flap motion (Fig. 37). For a stiff-inplane rotor, an instability was encountered at an advance ratio of about 0.4, just after the flap mode entered the 1/rev region due to the influence of the periodic coefficients (see Ref. 7). When this case was trimmed to zero hub moment, with fixed collective and shaft angles, it was stable, implying both that the instability is relatively weak (the analysis neglected structural damping) and that the stability solution is sensitive to the trim inforward flight (a reflection of the nonlinearity).

Reddy and Warmbrodt (Ref. 35) analyzed the flap-lag-torsion stability of an elastic blade in forward flight, including multi-blade coordinates and dynamic inflow. The equations were symbolically generated and coded by the computer, beginning with the formulation of Ref. 28 and the ordering scheme of Ref. 24. For the case of propulsive trim with a stiff-inplane rotor, using only the flap motion in the trim solution (as in Fig. 37) gave lower predicted stability than a full flap-lag-torsion trim solution (Fig. 38). Elastic flap/lag coupling (R) and the blade torsion motion significantly influenced the stability for a stiff-inplane rotor (Fig. 38). For a soft-inplane rotor, the stability was increased slightly using a full flap-lag-torsion trim solution, using flap/lag coupling ($R = 1$), or including the blade torsion motion; the character of the stability solution was not changed however. The effect of periodic coefficients was evident in the split roots around an advance ratio of 0.4 in Fig. 38. The periodic coefficients were not a major factor in the instability however; a constant coefficient approximation predicted the instability speed very well.

2.2.5 Bearingless Rotors

Bielawa (Ref. 36) developed an aeroelastic analysis (G400) for bearingless rotors, considering an elastic blade and redundant load paths at the root. A time-history solution gave steady state loads, and transients (for stability), in both hover and forward flight. A linearized analysis gave eigenvalues (for stability) in hover.

Bielawa, Cheney, and Novak (Ref. 37) conducted small-scale wind tunnel tests of a stiff-inplane composite bearingless rotor (CBR). The rotor used a flat flexbeam of carbon-epoxy, and a torque rod behind the flexbeam. A cantilever torque rod configuration (unconstrained at the root) showed significant pitch washout and pitch/flap coupling. With the torque rod pinned at each end, the rotor was stable and the measured performance and loads were similar to those of a hingeless rotor. A bearingless rotor was designed using an external torque tube, with a snubber at the root to minimize the couplings. Stability was calculated for soft-inplane and stiff-inplane configurations using the analysis of Ref. 36. The predictions showed a high-collective flap-lag instability, as for hingeless rotors with no pitch/lag coupling.

Hodges (Ref. 38) developed an analysis of bearingless rotor air resonance in hover (FLAIR, for Flexbeam Air Resonance). He considered rigid blades, attached to the hub by a single flexible beam or strap, with four rigid body degrees of freedom for the fuselage (excluding vertical and yaw motion, which are not coupled with the air-resonance modes for hover). Leading-edge or trailing-edge pitch-arm geometry was allowed, or both (a pitch control with snubber configuration). An iterative structural analysis obtained the trim flexbeam deflection, then numerical perturbations of the flexbeam stiffness gave the linear equations for a stability analysis.

Hodges (Ref. 39) compared results from the theory of Ref. 38 (for a single flexure with a rigid pitch arm and rigid blade) and from wind-tunnel measurements on a model bearingless rotor (with twin C-beams and

coupling (and even more in combination) was verified by the experiment (Fig. 18). Generally, the predictions were good, except that the damping measured at low collective with both pitch/lag and flap/lag coupling present was higher than predicted (Fig. 19). It was speculated that the discrepancy might be due to unsteady wake effects.

2.2.3 Elastic Blade Models

Hodges and Ormiston (Ref. 19) analyzed the elastic flap-lag-torsion motion of a uniform cantilever blade in hover, using the theory of Hodges and Dowell (Ref. 20). An axial-inplane-vertical-torsion deflection sequence was used; the torsion variable definition neglected a second order kinematic term however (except in the aerodynamic angle of attack). Later work showed that the number of modes used was probably not sufficient. Full flap/lag structural coupling was used in all cases. The effective pitch/lag and pitch/flap coupling due to torsion was discussed. The influence of the torsion degree-of-freedom was quasistatic above a torsion frequency of 4/rev, and negligible above a frequency of 10 to 15/rev. An instability was identified at low collective with moderate torsion stiffness and precone, produced by the effective pitch/lag coupling (positive, hence destabilizing for both soft and stiff inplane rotors with full flap/lag structural coupling).

Friedmann and Tong (Ref. 21) considered the elastic flap-lag stability of a hingeless rotor, neglecting structural flap/lag coupling. The motion was analyzed using an asymptotic expansion based on the method of multiple time scales (including an expansion for small advance ratio). For hover they obtained ellipse-like instability boundaries on the lag frequency-flap frequency plane (similar to Fig. 3, which is for a rigid blade). The asymptotic expansion method identified the instabilities as limit cycles, unstable on the top-right quadrant of the ellipse, and stable elsewhere.

Friedmann (Refs. 22 and 23) developed an analysis of hovering rotor blade flap-lag stability, including the rigid pitch degree of freedom (inboard of the bending flexibility). He neglected structural flap/lag coupling. The destabilizing influence of precone with low pitch frequency was shown (Ref. 22). The conclusions about the influence of droop were incorrect, because of a missing term in the equations.

Hodges and Ormiston (Ref. 24) extended the theoretical work of Ref. 19. They used six free vibration modes of the rotating blades, and considered zero or partial structural coupling (by setting the structural principal axes to a fraction of the aerodynamic pitch angle). Figure 20 shows the flap-lag stability boundaries for the elastic blade (compare to Fig. 4 for a rigid blade). Figures 21-23 show flap-lag-torsion stability boundaries as a function of lag and torsion frequencies, collective pitch, structural flap/lag coupling, and precone.

Friedmann and Straub (Ref. 25) developed a finite-element formulation for the analysis of elastic flap-lag stability in hover. They observed some influence of the second lag bending mode on the stability boundaries for flap/lag coupling around $R = 0.6$ (Fig. 24).

Hodges and Ormiston (Ref. 26) extended the analysis of Refs. 19 and 24 to include pitch-link flexibility and blade droop. The influence of the distribution of pitch flexibility between the pitch link and the blade elastic torsion was considered, including the role in the effective pitch/lag coupling. Figures 25 and 26 show the influence of f = ratio of torsional stiffness to pitch link stiffness ($f = 0$ for a torsionally rigid blade, $f = \infty$ for a rigid control system). There was little effect of the distribution when the precone and droop were both zero; and with all the flexibility in the blade torsion ($f = \infty$), precone and droop are equivalent. In general the effect of the distribution and the effect of droop are significant.

Friedmann (Ref. 27) developed a theory for flap-lag stability in hover, including both pitch link flexibility and elastic torsion. An axial-inplane-vertical-torsion deflection model was used, including structural flap/lag coupling. The analysis differed from Refs. 24 and 26 primarily in the representation of torsion deflection. The influence of the distribution of pitch flexibility between the pitch link and the blade elastic torsion was considered (Fig. 27). The effect of using the nonlinear equations for the static (trim) deflection was examined. The low collective instability due to precone was relatively weak, since 0.25% to 0.75% structural damping was sufficient to eliminate it (Fig. 28). Structural damping had little influence on the high collective boundary.

Kaza and Kvaternik (Ref. 28) derived the nonlinear equations for the dynamics of an elastic blade in forward flight. They considered both axial-inplane-vertical-torsion and axial-vertical-inplane-torsion deflection sequences in developing the equations.

Johnson (Ref. 29) developed an analysis of helicopter performance, loads, and stability (CAMRAD, for Comprehensive Analytical Model of Rotorcraft Aerodynamics and Dynamics). Warmbrodt and Peterson (Ref. 30) conducted a hover test of a full-scale hingeless rotor (BO-105). The lag damping measurements were compared with calculations obtained from the analysis of Ref. 29. It was essential to include in the analysis the blade pitch and torsion modes (because of their role in determining the effective pitch/lag coupling) and the dynamic inflow model for unsteady aerodynamics (Fig. 29). Good correlation was obtained for the influence of thrust (Fig. 30). A minimum of the measured damping around 400 rpm was not predicted however; the discrepancy could be caused by interaction with the wind-tunnel support (the regressing lag mode was in resonance with the longitudinal balance mode at 400 rpm), but including the body modes in the analysis did not change the predicted damping.

Lytwyn, Miao, and Woitsch (Ref. 9) developed an analysis of hingeless rotor air and ground resonance, using the rigid blade and hinge spring model. The analysis was applied to the BO-105 helicopter.

Burkam and Miao (Ref. 10) conducted a wind tunnel test of a model hingeless rotor with body pitch and roll motion (intended to model BO-105 air resonance; flap frequency 1.12/rev and lag frequency 0.62/rev, pitch bearing inboard of bending flexibility). They used the theory of Ref. 9 to identify favorable couplings, including the favorable pitch/lag coupling with under-precone. The original pitch links were stiff enough so that no instabilities were encountered in the test; the pitch link stiffness was significantly reduced in order to obtain measurable stability boundaries. The flap damping contributed to the body mode damping with this hingeless rotor. A high collective boundary was identified as air resonance, produced by the increase in roll mode frequency with collective increase. The analysis had shown that increased control system stiffness would stabilize this phenomenon, by reducing the effective couplings. A mild, low collective instability involved primarily the lag motion (although locking out the body increased the damping ratio by about 0.005). Figure 6 compares the predicted and measured stability boundaries. Figures 7-9 show the measured boundaries as a function of hub precone, inplane damping, and blade sweep.

Reichert and Huber (Ref. 11) analyzed the BO-105 helicopter with a rigid blade and hinge spring rotor model. A pitch-flap-lag-torsion hinge sequence was used for the flight dynamics calculations. A pitch-flap-lag rotor model with body roll and pitch motion was used for loads and stability calculations. The mechanism for pitch/flap and pitch/lag coupling in a hingeless rotor with an inboard feather bearing was discussed.

Huber (Ref. 12) analyzed the BO-105 helicopter using the method of Ref. 11, extended to five body degrees of freedom (no yaw motion or tail rotor model). The analysis was applicable to both hover and forward flight. Two-dimensional airfoil tables (including stall and compressibility effects) were used. The dynamic behavior was determined from a numerical integration in time for various control or external inputs. The pitch moment component of the flap moment when the blade lags, and the lag moment when the blade flaps, was discussed. Although the torsion moment at an arbitrary radial station was considered, the flap and lag moments were largest at the root and most of the torsion flexibility was in the control system. Hence, it was mainly the rigid pitch motion that was of concern. The influence of sweep, droop, control system stiffness, and thrust on the effective coupling was significant because of their influence on the elastic flap deflection (Fig. 10). Low precone was desired for increased lag stability through favorable pitch/lag coupling. The analysis showed a high-thrust instability due to stall. The reduction in lift-curve slope produced a loss of flap damping, which allowed a flap-lag instability (Fig. 11). This phenomenon was predicted to occur at about 2.2 g load factor in hover, hence was not within the operating envelope of the helicopter.

Miao and Huber (Ref. 13) conducted a wind tunnel test of a model soft-inplane hingeless rotor with body pitch and roll freedom. The theory of Ref. 12 was used, analyzing the predicted time histories to obtain frequencies and damping, just as for the experimental data. The analysis predicted the damping in hover well (Figs. 12 and 13).

Hansford and Simons (Ref. 14) developed an analysis for application to the Lynx hingeless rotor. They discussed the torsion moment due to the product of the flap and lag moments, noting that zero coupling was possible only with matched stiffness outboard of the feathering hinge or all the bending flexibility inboard of the feathering hinge (the latter is difficult to achieve except with hinges). Most of the torsion flexibility was was in the control system, so they only considered the pitch moment at the blade root.

2.2.2 Rigid Blade Models

Ormiston (Ref. 15) examined combinations of pitch/lag coupling and structural flap/lag coupling to stabilize soft-inplane hingeless rotors, using the theory of Ref. 8. Flap/lag coupling was introduced by means of pitch of the structural principal axes relative to the aerodynamic pitch. Pitch/lag and flap/lag coupling were predicted to be very stabilizing (Fig. 14).

Kaza and Kvaternik (Ref. 16) analyzed hingeless rotor flap-lag stability, considering the influence of hinge sequence in the rigid blade with springs model. A lag hinge inboard, flap hinge outboard sequence was used in Ref. 8. For the flap-then-lag sequence, additional aerodynamic forces are introduced, equivalent to a pitch/lag coupling equal to the blade coning angle (Fig. 15). They also considered the influence of forward flight.

Ormiston and Bousman (Ref. 17) conducted a test of the flap-lag stability of a hingeless model rotor in hover. The blades used hinges and flexures. The experiment was designed to check the predictions of Ref. 8: the minimum stability with equal flap and lag frequencies (without flap/lag coupling); and the significant stability increase with structural flap/lag coupling. Figure 16 shows the capability of the theory. In addition, a stall-induced flap/lag instability was found at high thrust, attributed to the reduction of the flap damping because of the reduced lift-curve slope in the stall regime (Fig. 17).

Bousman, Sharpe, and Ormiston (Ref. 18) conducted a hover test of a model soft-inplane hingeless rotor, in order to verify the theory of Ref. 8. A rigid blade with flap and lag hinge flexures was used. Pitch/lag coupling was introduced by skewing the lag flexure, while flap/lag coupling was introduced by pitching the principal axes of the flexure relative to the hub plane (the blade collective pitch was changed outboard of the flexures). The significant stability improvement with pitch/lag or flap/lag

where M_x and M_z are the section flap and lag bending moments; EI_β and EI_ζ are the flap and lag bending stiffnesses; z'' and x'' are the out-of-plane and inplane curvatures. Hence there is a nonlinear load coupling bending and torsion, proportional to the difference between the bending stiffnesses. This coupling is zero for a matched stiffness design. Normally the chordwise stiffness is much greater than the flapwise stiffness, but nearly equal stiffnesses can be obtained at the root of a soft-inplane rotor blade.

2.1.3 Ground and Air Resonance

When the aircraft body motion is added to the problem, the stability phenomena that are often of most importance are ground resonance and air resonance. Ground resonance is a dynamic instability involving the coupling of the blade lag motion with the inplane motion of the rotor hub (see Ref. 7 for a complete discussion and analysis). This instability is characterized by a resonance of the frequency of the rotor lag motion (specifically the regressing lag mode in the nonrotating frame) and a natural frequency of the structure supporting the rotor (Fig. 2). Since the lag frequency depends on the rotor rotational speed, such resonances define critical speed ranges for the rotor. An instability is possible at a resonance if the rotating lag frequency is below 1/rev, as it is for articulated and soft inplane hingeless rotors. With articulated rotors, the critical mode is usually an oscillation of the helicopter on the landing gear when in contact with the ground. The classical ground resonance analysis considers four degrees of freedom: longitudinal and lateral inplane motion of the rotor hub, and the progressing and regressing lag modes. The actual vibration modes of the rotor support, such as the motion of the helicopter on its landing gear, will probably involve tilt of the shaft as well, but it is the inplane hub motion that dominates the ground resonance phenomenon, particularly for articulated rotors. Also for articulated rotors, the damping of the rotor and support comes almost entirely from mechanical dampers and structural damping, so the aerodynamic forces are neglected. The coupling of the body and the rotor lag motion is determined by the first moment of inertia of the blade. For small rotor mass compared to body mode generalized mass (which is usually the case), the Deutsch criterion (Ref. 7) provides a good estimate of the damping required for stability:

$$\frac{C_x}{\omega_x^2} C_\zeta > \frac{N}{4} \frac{1 - \nu_\zeta}{\nu_\zeta} S_\zeta^2$$

at a rotor speed of $\Omega = \omega_x/(1 - \nu_\zeta)$. Here C_x and C_ζ are the dimensional body mode and rotor lag damping coefficients; ω_x is the frequency (rad/sec) of the body mode; N is the number of blades; ν_ζ is the per-rev natural frequency of the lag motion; and S_ζ is the first moment of inertia of the blade (product of the blade mass and the radial center of gravity location). The factor $(1 - \nu_\zeta)/\nu_\zeta$ determines the severity of the instability. For an articulated rotor this factor is large, and a mechanical lag damper is needed. For a soft-inplane hingeless rotor this factor is small, so the blade structural damping may be sufficient for stability. For a stiff-inplane hingeless rotor this factor is negative, so there is no ground resonance problem.

A similar stability phenomenon exists in flight, particularly with a hingeless rotor, and then it is called air resonance. The blade flap motion and the rotor aerodynamics must be included in an analysis of air resonance, since the flap stiffness and aerodynamics determine the frequency and damping of the body modes in flight. The critical stability case still occurs at a resonance with the regressing lag mode (Fig. 2). For air resonance, there are no springs on the body motion, as exist in the ground resonance problem. To the rotor degrees-of-freedom the analysis adds the aircraft rigid body motions, hence the eigenvalues associated with the flight dynamics. Singly, the pitch or roll motion would each add a real, damped eigenvalue; together they add an oscillatory mode (if the pitch and roll inertias are not too different). The pitch, roll, longitudinal, and lateral motions together add two real, damped roots and two unstable or low-damped oscillatory modes (in hover). When the frequencies of these modes are low, they are not part of the air resonance problem. Rather, air resonance involves the regressing flap mode, which includes considerable body motion, particularly with a hingeless rotor. When the frequencies of the body modes are high and the frequency of the regressing flap mode low, the above distinctions are less useful. The coupled flap-lag motion has reduced stability at high collective, hence air resonance stability tends to decrease as collective pitch increases. Articulated rotors have mechanical lag dampers and relatively little body motion in the regressing flap mode. Thus air resonance is primarily a problem for soft-inplane hingeless or bearingless rotors.

2.2 Stability--Recent Developments

2.2.1 Hingeless Rotors

Ormiston and Hodges (Ref. 8) developed a theory for flap-lag stability, based on a rotor blade model consisting of rigid blades with hinge springs. They identified the stabilizing influence of the proper choice of pitch/lag coupling, and the use of structural flap/lag coupling. Their analysis provided a description of the basic high collective flap-lag instability, including a definition of the critical case of flap frequency and lag frequency equal to 1.15. Figures 3 to 5 show the flap-lag stability boundaries as a function of flap and lag frequencies, collective pitch, structural flap/lag coupling ($R = 0$ for no coupling, $R = 1$ for complete coupling), and pitch/lag coupling.

$$K_t\phi_t = \beta_e M_\zeta - \zeta_e M_\beta$$

$$K_c\phi_c = (\beta_e - \beta_d)M_\zeta - (\zeta_e + \zeta_s)M_\beta$$

The pitch angles, precone, droop and sweep, elastic flap and lag deflection, and flap and lag moments are defined in Fig. 1. The total pitch deflection is then:

$$\theta = \phi_t + \phi_c = \left(\frac{1}{K_c} + \frac{1}{K_t}\right)(\beta_e M_\zeta - \zeta_e M_\beta) + \frac{1}{K_c}(-\beta_d M_\zeta - \zeta_s M_\beta)$$

$$= \frac{1}{K_\theta}(K_\zeta - K_\beta)\beta_e\zeta_e + \frac{1}{K_c}(-K_\zeta\beta_d\zeta_e - K_\beta\zeta_s\beta_e)$$

where the K's are the spring constants, and $K_\theta^{-1} = K_c^{-1} + K_t^{-1}$. Hence, because of the blade pitch flexibility, bending of the blade produces a pitch deflection. With no droop or sweep, the pitch moment is:

$$M_\theta = K_\theta\theta = I\Omega^2(1 + \nu_\zeta^2 - \nu_\beta^2)\beta_e\zeta_e$$

where the flap and lag springs have been written in terms of the rotating natural frequencies (per rev; I is the blade moment of inertia and Ω is the rotor rotational speed). The effective pitch/lag coupling (positive for lag back, pitch down) is:

$$K_{P_\zeta} = -\frac{\partial\theta}{\partial\zeta_e} = -\frac{K_\zeta - K_\beta}{K_\theta}\beta_e + \frac{K_\zeta}{K_c}\beta_d$$

$$= -\frac{K_\zeta - K_\beta}{K_\theta}\,\frac{\beta_I - \beta_p + \beta_d}{\nu_\beta^2} + \frac{K_\zeta}{K_c}\beta_d$$

where ν_β is the flap frequency (per rev), and the ideal precone in hover (the coning angle for a flap frequency of exactly 1/rev; see Ref. 7) is:

$$\beta_I = \frac{\gamma}{8}\frac{6C_T}{\sigma a}$$

(γ = Lock number, a = two-dimensional lift curve slope).

With a matched stiffness design (equal flap and lag hinge springs), or ideal precone so the elastic flap deflection is zero, the first term in the equation for the pitch/lag coupling is absent and the coupling is solely due to droop producing a moment about the control system spring. In general, the first term is important also. In particular, at low pitch the precone will be larger than ideal, hence the elastic flap deflection β_e will be negative, and from above the pitch/lag coupling will be positive. It follows there can be an instability at low collective for a soft-inplane rotor with precone, because of the effective pitch/lag coupling introduced by the blade pitch flexibility. Alternatively, using a precone value less than the ideal precone, or increasing the blade pitch stiffness, will be favorable for stability. The damping and inertia of the torsion motion are much less important. It is found that a quasistatic torsion model is adequate, except for low torsion stiffness; but a model without the torsion motion entirely is not adequate except for very high torsion stiffness.

Similarly the effective pitch/flap coupling (positive for flap up, pitch down; $K_{P_\beta} = \tan\delta_3$) is:

$$K_{P_\beta} = -\frac{\partial\theta}{\partial\beta_e} = -\frac{K_\zeta - K_\beta}{K_\theta}\zeta_e + \frac{K_\beta}{K_c}\zeta_s$$

$$= -\frac{K_\zeta - K_\beta}{K_\theta}\,\frac{\gamma C_Q/\sigma a}{\nu_\zeta^2} + \frac{K_\beta}{K_c}\zeta_s$$

(ν_ζ = lag frequency, per rev; C_Q = torque coefficient). Note that with neither droop nor sweep, the pitch/lag or pitch/flap coupling per radian of elastic blade deflection is:

$$K_P \text{ per rad} = -\frac{I\Omega^2}{K_\theta}(1 + \nu_\zeta^2 - \nu_\beta^2)$$

A similar result can be derived for the torsion moment at an arbitrary radial station on the blade (Ref. 7). Considering the out-of-plane forces with a moment arm due to inplane deflection, and the inplane forces with a moment arm due to out-of-plane deflection, gives a torsion moment:

$$\Delta T = M_x x'' - M_z z'' = (EI_\beta - EI_\zeta)x''z''$$

spanwise, y inplane, and z vertical). For vertical laminates the plane of symmetry was vertical (the x-z plane). Hence, the shear stress, σ_{xy} or σ_{xz} for the horizontal or vertical laminates, respectively, were coupled to the normal strain ϵ_{xx} by the parameter Q_{16}. Transforming the orthotropic material properties at ply angle Λ to the section coordinate system gave Q_{16} for each laminate. Q_{16} would be zero for isotropic materials, or with ply angles of $\Lambda = 0$ or 90°. A nonzero value of Q_{16} introduced linear and nonlinear terms into the equations for beam deflection, extension, and torsion, producing bending/torsion and extension/torsion coupling. For a symmetric orientation of the plies on the sides of the spar, the ply angle introduced a pitch/lag type of coupling, that had a significant effect on the lag damping (Fig. 88). For a symmetric orientation of the plies on the top and bottom of the spar, the ply angle introduced a pitch/flap type of coupling, that had a significant effect on the flap mode frequency. For an antisymmetric orientation of the plies, the ply angle introduced an extension/torsion coupling; this was a nonlinear effect, but had a significant influence on the stability (Fig. 89).

4. DYNAMICS OF ROTORCRAFT CONFIGURATIONS

Finally, the dynamics of various rotorcraft configurations are considered: hingeless rotors, bearingless rotors, rotors utilizing circulation control, coupled rotor/engine dynamics, articulated rotors, and tilting proprotor aircraft. The emphasis is on describing the design approaches, problems encountered during development, and solutions to those problems.

4.1 Hingeless Rotors

The hingeless rotor replaces the flap and lag hinges of the articulated rotor with bending flexibility at the blade root. The pitch bearing is retained. The hingeless rotor offers the advantages of mechanical simplicity and increased hub moment capability. The latter has a favorable influence on handling qualities, by increasing the damping and control power of the rotor. These advantages are accompanied by new stability phenomena, and some adverse effects of the increased hub moments, including higher vibration and gust response, and increased angle-of-attack instability. The dynamics analyses required to support the design of a hingeless rotor are more complicated, since structural modes replace the fundamental rigid body modes of the articulated blade.

Ormiston (Ref. 5) and Strehlow and Enenkl (Ref. 98) summarized the design considerations for hingeless rotors. The frequencies of the fundamental flap and lag modes are the first design choices. A flap frequency of 1.10 to 1.15/rev was typical of the first successful hingeless rotors. The current trend is to require lower flap frequencies, for reduced vibration and gust response and to minimize adverse handling qualities effects at high speed. The goal is a flap frequency in the range 1.06 to 1.08/rev (an articulated rotor would have a frequency less than about 1.04/rev). This range is difficult to achieve with a hingeless rotor, although the use of small hubs made from composite materials helps. A soft-inplane rotor (lag frequency below 1/rev) will be susceptible to air and ground resonance instabilities, and hence may require a lag damper. A lag frequency above 0.6/rev is desired for air and ground resonance stability, and the frequency must be below about 0.8/rev for acceptable loads. A matched stiffness design would require a lag frequency of about 0.5/rev. A stiff-inplane rotor (lag frequency above 1/rev) has no ground or air resonance problems, but will have higher loads, and the flap-lag-torsion stability phenomena generally display a greater complexity and sensitivity. Acceptable loads and strength have been achieved by the use of advanced materials, and most often by the selection of the soft-inplane configuration for main rotors. Acceptable stability can be achieved by designing the rotor for minimum coupling of the blade modes (such as by using nearly a matched stiffness design); or by designing the rotor specifically for favorable values of pitch/lag and flap/lag coupling over the operating range.

4.1.1 AH-56A

Carlson and Kerr (Ref. 58) described the design of the AH-56A helicopter: a four-bladed, hingeless, gyro-controlled rotor. The rotor was stiff-inplane, with a lag frequency of about 1.3/rev. The control gyro utilized feather-moment feedback with a swept-forward blade to improve the aircraft dynamics. The dynamic characteristics were analyzed using the REXOR code, which produced a time history solution. The rotor was represented by two flapwise and one inplane bending modes, the control system flexibility, and quasi-static torsion motion; the body and gyro motion included pitch, roll, vertical, and rotational speed degrees of freedom. Figure 90 compares the measured and calculated stability.

Anderson (Ref. 99) described a reactionless flap-lag instability that was encountered in AH-56A flight tests at low speed (20 to 30 knots) and high lift, producing chordwise loads sufficient to buckle the blade trailing edge (Fig. 91). The blade droop contribution to the effective pitch/lag coupling was identified. The stability problem was cured by increasing the blade droop (producing negative pitch/lag coupling, which is stabilizing for a stiff-inplane rotor with full flap/lag structural coupling).

Anderson and Johnston (Ref. 100) described a phenomenon (called a hop mode) encountered on the AH-56A, involving coupling of the regressive lag mode, the body roll mode, and the rotor coning mode. There was a coupling and coalescence of the coning mode frequency with the roll and lag modes as aircraft forward speed increased. An instability occurred at about 200 knots (Fig. 92). The cure involved reducing the kinematic pitch/flap coupling and increasing the control system stiffness, so that the frequencies of these modes would not vary with forward speed; the instability boundary was thereby increased to about 270 knots.

4.1.2 BO-105

Huber (Ref. 12) described the BO-105 helicopter: a soft-inplane, hingeless rotor. The rotor had a stiff titanium hub, incorporating the pitch bearings, and fiberglass blades. Fiberglass was used to achieve low stiffness and good fatigue life. All the blade bending occurred outboard of the pitch bearings. The fundamental design approach was to use the strong couplings inherent in such a rotor to provide good dynamic characteristics and stability (Fig. 93). The rotor had no lag damper. The rotor had no droop or sweep, 2.5° precone, a flap frequency of 1.12/rev, lag frequency of 0.67/rev, and pitch frequency of 3.6/rev. Hence, the effective pitch/lag and pitch/flap couplings were about 0.1 per degree of elastic blade deflection (Fig. 10). The analysis of Ref. 11 was used to support the design.

Reichert (Ref. 57) and Reichert and Weiland (Ref. 101) discussed the BO-105 rotor loads. The maximum oscillatory bending moments normally occurred at the blade midspan on an articulated rotor, but occurred at the blade root on a hingeless rotor. The hingeless rotor peak loads (at the root) were much higher than the articulated rotor peak loads; but on the outboard portion of the blade the loads were lower than on an articulated rotor. The 1/rev blade motion dominated the flap and lag bending moments. Because the blade loads were dominated by the 1/rev motion of the fundamental flap and lag modes, good correlation between predicted and measured loads was obtained (using the analysis of Ref. 11; see Fig. 55). The aeroelastic couplings were important for the loads as well as for stability; hence, the blade pitch motion must be included in the analysis. For helicopter vibration, the higher harmonics and additional modes were important, so a better analysis than that of Ref. 11 would be required for good predictions.

Reichert and Weiland (Ref. 101) discussed the BO-105 helicopter ground and air resonance characteristics. The relatively high lag frequency and lag damping were sufficient to preclude any stability problem, without a mechanical lag damper.

Kloppel, Kampa, and Isselhorst (Ref. 102) presented measurements of the lag damping of the BO-105 rotor in hover on a whirl tower. Warmbrodt and Peterson (Ref. 30) measured the damping of the full-scale rotor on a wind tunnel test stand. Figure 94 compares the measurements with the calculations using the analysis of Ref. 29.

Strehlow and Enenkl (Ref. 98) identified the source of the BO-105 blade lag damping as primarily mechanical losses in the blade root attachment fitting. Consequently, the equivalent viscous damping was a nonlinear function of the blade lag bending moment (Fig. 95). Kloppel, Kampa, and Isselhorst (Ref. 102) showed the influence of the nonlinear structural damping on the calculated forward flight stability (Fig. 96; the lower theoretical curve corresponds to Fig. 95, while the upper curve is for a slightly different variation of structural damping with lag moment).

4.1.3 Lynx

Balmford (Ref. 103) described the development of a research hingeless rotor on a Scout helicopter. The intent was to match articulated rotor behavior by minimizing the structural flap/lag/torsion coupling and minimizing the feather moments due to flap and lag motion. Hence, the hub configuration consisted of an inboard flap flexure, then a feathering bearing (so there would be no feather moment due to flapping), then a matched stiffness lag flexure outboard. The rotor had a high control system and blade torsion stiffness. The compromise between 1/rev blade loads and ground/air resonance stability led to a lag frequency of 0.64/rev. A ground resonance instability was encountered in tests at maximum overspeed rotor rpm, due to lower body frequency and lower lag damping than anticipated. Therefore, lag dampers were installed. Air resonance (analyzed for hover only) and vibration were no problem. Rotor bending loads were calculated using a normal mode method, and good correlation with flight test results was achieved (Fig. 54). Control loads were no problem.

Hansford and Simons (Ref. 14) described the design of the Lynx hingeless rotor. A hingeless hub was desired for simplicity, but dynamic characteristics not too far from those of an articulated rotor were preferred. Rather than use the blade couplings to control the rotor dynamics, and deal with the adverse effects of the couplings and sensitivity to flight condition, the couplings were minimized throughout the flight envelope. A rotor designed with an inboard feather bearing, a flap frequency of 1.09/rev, lag frequency of 0.58/rev, and pitch frequency of 5/rev would have an effective pitch/lag and pitch/flap coupling of 0.4 per degree of elastic deflection. For a flap frequency in the range 1.09 to 1.14/rev, zero coupling would require a lag frequency of 0.43 to 0.55/rev--too low for available materials, and too low for good ground resonance stability. Thus the design approach for the Lynx concentrated on matching the blade flap and lag stiffness where the product of the bending moments was highest at the root (Fig. 97). This was accomplished by using a circular, flexible element outboard of the pitch bearing. A fully matched stiffness design would require that this flexible element be too long, and the lag frequency would be too low. Hence, a flap flexure was introduced between the hub and the pitch bearing, to reduce the effective coupling associated with out-of-plane deflection relative to the feathering axis (this design also allowed independent selection of the flap and lag frequencies). Finally, for better ground resonance stability, the lag frequency was increased to 0.64/rev and a lag damper was used. The resulting design had only a small effective pitch/lag and pitch/flap coupling: 0.015 per degree of elastic deflection.

Berrington (Ref. 104) discussed the Lynx rotor design and dynamic characteristics. There was no problem with ground or air resonance. The vibration was initially high, but was reduced to 0.05 to 0.10 g by manipulation of the airframe structural modes.

4.1.4 ABC

Burgess (Ref. 105) describes the Advancing Blade Concept (ABC) helicopter: a stiff-inplane hingeless rotor. The hub moment capability of a hingeless rotor was used in the concept to allow the rotor to fly with a net rolling moment in forward flight, alleviating the retreating-blade stall limit. To balance the rolling moment, a coaxial-rotor configuration was used, and adequate blade clearance between the two rotors in forward flight required high stiffness. The ABC rotor flap frequency was about 1.5/rev and the lag frequency about 1.4/rev. Blade loads for design of the rotor were calculated using a normal mode analysis. Flutter analysis for the rotor design was performed using a frozen coefficient method, considering only flap and torsion modes.

Young and Simon (Ref. 106) discussed ABC helicopter dynamics. The high stiffness required for tip clearance resulted in good stress margins in the blade, but also produced high bending loads through the feather bearing at the blade root, with a significant impact on fatigue life. In flight tests the shaft stresses exceeded endurance in descent, because of the hub moment needed to balance the horizontal tail moment; this problem was corrected by introducing coupling of the elevator to the collective stick. Blade inplane stability was no problem. The coaxial rotors were phased such that the 3/rev symmetric vibratory forces (vertical force, longitudinal force, and pitch moment) tended to cancel. The vibration due to 3/rev lateral force and roll moment was high, however. The vibration was significantly reduced by an absorber (Fig. 98).

Abbe, Blackwell, and Jenney (Ref. 107) discussed ABC rotor stability. As a result of the high stiffness and coaxial configuration, the lag mode was almost pure inplane motion, involving little coupling with the flap, torsion, control, or airframe motion. The measured damping showed little variation with airspeed (Fig. 99), but the upper rotor damping did decrease for high rates of descent at 80 to 100 knots. A normal mode analysis with time integration was used to calculate the stability.

Linden and Ruddell (Ref. 108) and Ruddell et al. (Ref. 109) discussed the ABC helicopter vibration characteristics. With the coaxial configuration, the vibration depended on the rotor phasing: for a blade crossover at 90° azimuth, the symmetric hub forces (vertical force, longitudinal force, pitch moment) tended to cancel; for a blade crossover at 0° azimuth, the antisymmetric hub forces (lateral force, roll moment, yaw moment) tended to cancel. The dominant excitation of the airframe came from the pitch or roll moment, because of the high flap stiffness. The flight tests were first conducted with a 90° crossover; excessive cockpit vibration was encountered, and the maximum speed achieved was 204 knots. Then the flight tests were conducted with a 0° crossover, and the vibration was significantly reduced (Fig. 100). The vibration level was still high, but no vibration treatment had been installed yet. It was established that the vibration level of the ABC tended to be lower than that of a conventional helicopter at the same speed (comparing both without vibration treatment); but the ABC vibration was higher than that of a conventional helicopter at their respective maximum speeds. The maximum speed achieved in the flight tests was 263 knots (diving), limited by upper rotor shaft endurance loads. A possible correction for this loads limit would involve using the elevator to reduce the hub moment required from the rotor.

4.1.5 BK-117

Huber and Masue (Ref. 110) described the design of the BK-117 helicopter: a soft-inplane hingeless rotor. The design philosophy and the resulting rotor configuration were the same as for the BO-105 rotor: a stiff titanium hub with pitch bearings, and fiberglass blades. The flap frequency was 1.10/rev and lag frequency 0.65/rev (compared to 1.12 and 0.67 for the BO-105). As a result of the BO-105 analytical work (Ref. 11), the blade center of gravity was placed at 23.5% chord (compared to 25% chord for the BO-105) in order to introduce blade center-of-gravity/aerodynamic-center coupling favorable for handling qualities. The blade had 2.5° precone and no droop (same as BO-105), and 1.0° aft sweep (the BO-105 had no sweep). Whirl tower tests showed stability somewhat better than the BO-105 (Fig. 101; the analytical results correlated best for the BO-105, undoubtedly reflecting the long use of the analysis for that rotor). Flight tests showed that the loads and air/ground resonance stability were no problem (Fig. 102). The vibration level was still moderately high after just tuning the blade frequencies (Fig. 103). A NASTRAN analysis and shake test identified fuselage modes near 4/rev; local stiffening of the structure showed a decrease in local vibration but only a minor change in cabin vibration. Flap pendulum absorbers on the blades (originally demonstrated for the BO-105) significantly reduced vibration in transition. A multiaxis antiresonance isolation system, involving four vertical mechanical isolators and one lateral hydraulic isolator, was very effective in reducing the vibration at all speeds, even during transients.

Strehlow and Enenkl (Ref. 98) discussed the BK-117 design philosophy and development. The rotor precone resulted in upward elastic flap deflection of about 0.9° in hover, hence pitch/lag coupling of -0.2, which was favorable for flap-lag stability. The BK-117 used blade tuning weights to control 3/rev and 5/rev rotor loads and 4/rev hub moments, hence to minimize vibration. Ground resonance stability was not a problem. The aircraft had a stiff landing gear, so the body pitch mode had the lowest frequency and the roll mode resonance was above normal rotor operting speed.

4.2 Bearingless Rotors

In the quest for design simplicity, the next logical step from the hingeless rotor is the bearingless rotor, in which structural flexibility rather than hinges and bearings is used to provide blade pitch as well as flap-lag motion. Such a hub configuration becomes practical largely through the use of composite

materials. Simplicity is a goal because of the favorable implications for rotor system weight, cost, and reliability. The elimination of the feather bearing, however, introduces even more complicated dynamic phenomena than for the hingeless rotor. Bousman, Ormiston, and Mirick (Ref. 111) and Strehlow and Enenkl (Ref. 98) discussed design considerations for bearingless rotors. As for hingeless rotors, it is desired to have a low flap frequency in order to minimize gust response, vibration, and adverse handling qualities effects. Bousman gives a flap frequency goal of about 1.03 to 1.05/rev; Strehlow defines the goal as 1.06 to 1.08/rev. These values can be achieved by introducing a structural flap flexure into the design, which is possible with composite materials. Bearingless rotor designs for main rotors are soft-inplane, for manageable blade loads. Generally some lag damping source beyond structural damping is desired to improve aeromechanical stability. Most bearingless tail rotor designs are stiff-inplane. Many designs are being developed and tested. Perhaps the most common configuration now is a flexbeam with an inboard flap flexure, an external torque tube, and a snubber/damper at the root of the torque tube.

4.2.1 XH-51A

Donham, Cardinale, and Sachs (Ref. 112) described the development of a soft-inplane bearingless rotor for the XH-51A helicopter. The rotor used steel flexures at the root, with polar symmetry for a matched stiffness configuration; the lag frequency was 0.65/rev. The low inplane stiffness was necessary to achieve the desired torsion flexibility. Pitch control was by means of a steel torque rod forward of the flexbeam, mounted with flexible couplings to eliminate bending loads. The XH-51A rotor was gyro-controlled, although a smaller gyro was needed compared to the stiff-inplane hingeless rotor. The matched stiffness eliminated feather moments due to flap or lag deflection, which would be undesirable feedback signals to the gyro; a smaller and simpler control system was thereby possible. The rotor system was 11% lighter than the stiff-inplane hingeless rotor. In flight tests (Fig. 104) the aircraft showed marginal air resonance stability: an instability at about 86% normal rotor speed, which was considered an insufficient margin for autorotation (design operating range was 89% to 106% rpm). The rotor was tested with negative pitch/flap and pitch/lag coupling; analysis suggested that positive pitch/flap coupling would stabilize the air resonance. Ground resonance stability was acceptable (critical rotor speeds were above 106% rpm) on a smooth, prepared surface with complete contact of the skids and the ground. Partial skid contact, on a rough or soft surface, could have resulted in an unstable condition.

4.2.2 BMR

Staley, Gabel, and MacDonald (Ref. 113) described the development of the Bearingless Main Rotor (BMR). The soft-inplane rotor used twin fiberglass flexbeams, extending to 25% radius, and a graphite torque rod between the beams, cantilevered at the outboard end and pinned at the root. The rotor was tested on a BO-105 aircraft, and the fundamental frequencies were chosen to match those of the BO-105 hingeless rotor: flap frequency 1.12/rev and lag frequency 0.69/rev. The flexbeams used 12.5° prepitch to introduce structural flap/lag coupling, and 2.5° negative droop to improve stability. The rotor design was developed using a hover stability analysis (based on the rigid blade and hinge spring model with prescribed couplings), and small scale wind tunnel tests. Flight tests showed that lag damping and air resonance were no problem (Figs. 105 and 106). Vibration characteristics were similar to those of the BO-105. The BMR air/ground resonance damping was generally lower than that for the BO-105; the structural damping was about 1% for the BMR compared to 3% for the BO-105 (Fig. 107). Ground resonance tests with the original landing gear configuration showed an instability at low collective, at 102% rpm on concrete and 95% rpm on turf (the body frequencies are lower on turf; Figs. 108 and 109). When the landing gear skid was stiffened, the ground resonance stability on concrete was acceptable. Neutral stability was then encountered on turf at low collective and 97.5% rpm. For these flight tests it was possible to simply avoid that operating condition; for a new helicopter a soft landing gear design would be used to eliminate the problem. The analysis was not able to predict all of the ground resonance problems; it assumed complete contact of the landing gear with the ground, and did not allow for variations of landing gear characteristics with rotor thrust.

Dixon (Ref. 114) described the development of the BMR design. The design started with an I-beam of Kevlar for the flexbeam (for low stress, low torsion moment due to twist, and ease of fabrication); and a leading edge torque rod. A torque sleeve was rejected because it would be necessary to develop an elastomeric bearing for the inboard attachment, and because no fairing was desired during the flight research (to allow inspection and instrumentation maintenance). Concern about lack of fatigue data and the compressive strength of the Kevlar led to the use of S-glass for the flexbeam; the stress was lower with the S-glass, but the torsion moment needed to twist the blade was higher. The outboard connection was simplified by splitting the I-beam into two C-beams, and placing the torque rod between the C-beams, at the center of twist. This design introduced the difficulty that a dual beam is not a classical problem in structural and dynamics analyses. Graphite for the torque rod provided the simplest and lightest design. Separate beams for each blade, rather than a through-hub design, was chosen to allow research variations in flexbeam configuration. Negative droop of 2.5° relative to the torsion flexure was used for stability. The wind tunnel model tests showed that 12.5° of beam pretwist improved the stability, although it complicated the hub design. Limits in the analytical tools for stability prediction included the use of an equivalent hinge model, inadequate model of the landing gear, and no forward flight. The analysis of Ref. 40 was developed in response to these limitations. Limits for loads predictions included the lack of a true multi-load path model.

Warmbrodt, McCloud, Sheffler, and Staley (Ref. 41) conducted a full-scale wind tunnel test of the BMR rotor. The measured damping compared well with flight test results (Fig. 110), indicating the absence of

coupling with body motions at normal rotor speed. Sheffler, Warmbrodt, and Staley (Ref. 115) considered lag damping augmentation in the full-scale wind tunnel test of the BMR (Ref. 41). An elastomeric damping material was bonded to the top and bottom surfaces of the C-beams, and constrained by an outer layer of graphite reinforced epoxy laminate. The lag damping of the rotor increased by 1.5% critical (about 50% higher); the lag frequency was increased by about 0.04/rev.

Warmbrodt and Peterson (Ref. 30) compared the hover stability measured in full-scale tests of the BMR and the BO-105 rotors. At design rotational speed and thrust, the BMR was more stable. The BMR stability was lower than that of the BO-105 at low thrust, and significantly lower at 82% rotor speed.

McHugh, Staley, and Sheffler (Ref. 116) conducted model wind-tunnel tests to develop a bearingless rotor with low flap frequency. The goal was a flap frequency of about 1.04/rev (compared to 1.12/rev for the BMR). Two designs were considered: a dual beam configuration, like the BMR, with a flap frequency of 1.03 to 1.05/rev (for zero to design thrust, respectively); and a single flexstrap configuration (torque rod below the strap), with a flap frequency of 1.03 to 1.04/rev. Air resonance instabilities were encountered below 100% rpm (Figs. 111 and 112). Adding constrained layers of elastomeric damping material significantly increased the damping (Fig. 112); reducing the lag frequency lowered the rotor speed of the instability. The same stability boundary and damping levels as the BMR were achieved for the dual beam configuration with a lag frequency reduced to 0.58/rev; and for the single flexstrap configuration with a lag frequency of 0.62/rev and the added damping material (Fig. 113).

4.2.3 Triflex

Cassier (Ref. 117) described the development of the Triflex main rotor: a three-bladed, soft-inplane bearingless rotor, tested on a Gazelle helicopter. The Triflex had a single beam at each blade root with pitch, flap, and lag flexibility; and a rigid pitch horn at the outboard end of the flexure. Each flexure was constructed of unidirectional glass-fiber and epoxy-resin rovings embedded in an elastomeric matrix. The elastomeric matrix maintained the spacing between the rovings during bending, and provided damping. The rotor had a flap frequency of 1.06/rev, lag frequency of 0.72/rev, 2.5° precone, and pitch/flap coupling of 0.5 for stability. Whirl tests showed no stability problems, but the lag response was high during start and stop at low collective. The lag damping was about 1%, which was less than predicted. Flight tests showed a weak tendency for a ground resonance instability (with a resonance slightly above normal rotor speed) because of the low lag damping. The problem was cured by installation of a hydraulic damper on the landing gear. There was no air resonance stability problem, but a resonance of the regressing lag mode with the engine lateral mode at about 110% rpm resulted in increased vibration, particularly at high speed. The problem was cured by locking out the flexible longitudinal mount of the main gear box. Generally the vibration (normally a concern with the Gazelle helicopter because of a fuselage mode near 3/rev) was increased with the Triflex rotor. Maximum forward speed was achieved after installation of bifilar pendulums. The control system loads were higher than for an articulated rotor. The flight envelope was therefore limited somewhat by control loads, since the normal Gazelle control actuators were used.

In further development of the Triflex rotor, Aerospatiale increased the number of blades to four, in order to reduce the viration with the Gazelle fuselage. The four-bladed hub was also easier to fabricate. The elastomer provided more lag damping than for the three-bladed hub, but a more conservative flight test approach required the installation of a lag damper to insure ground resonance stability. It is anticipated that the use of a new elastomeric matrix will eliminate the need for a lag damper. Flight tests showed no pitch-up tendency of the aircraft, acceptable vibration (without absorbers), and no stability problems. Primary development of the Triflex hub configuration was completed. Some design changes would be desirable, particularly to improve fatigue life: a new elastomer, stronger control actuators, and stiffer pitch arms.

4.2.4 Model 680

Metzger (Ref. 118) described the Model 680: a soft-inplane bearingless main rotor. The rotor was developed with the goals of reducing the number of parts by 50% and the weight by 15%, increasing the fatigue life, and achieving low vibration with minimum weight penalty. The four-bladed hub had flexbeams with a flap flexure inboard and a torsion section outboard. For simplicity, the initial design had a pitch horn attached to the blade at the end of the flexbeam (20% radius), with no shear restraint or lag damper. To eliminate the large moments at the blade interface produced by control input and a significant pitch control washout, the design was changed to a torque tube with shear restraint. In model tests of configurations without dampers, the stability margin was not acceptable. Hence, elastomeric lag dampers were added at the shear restraints. The flexbeam and torque tube were made from fiberglass-epoxy. The torque tube was stiffness designed, so graphite-epoxy would be lighter. The rotor and pylon were designed for low vibration: the rotor dynamics were tailored to reduce 4/rev vibration; a linkage-focused pylon with longitudinal and lateral restraint springs was used; and vertical isolation was achieved using "Liquid Inertial Vibration Eliminators" between the transmission and pylon. The rotor was flight tested on a Model 222 helicopter. Shake tests showed 3% rotor damping would be needed for ground resonance stability; at least 3.5% was available from the lag dampers alone. Ground and air resonance were no problem, and the loads measured in flight indicated a fatigue life of at least 10,000 hr for the hub. The 4/rev vibration was below 0.1 g from hover to 170 knots. The vertical isolators were not needed at high speed, but were responsible for eliminating a transition vibration peak of 0.3 g at 30 knots.

Weller (Ref. 119) conducted hover and wind tunnel tests of a model of the 680 rotor: a soft-inplane bearingless rotor. The rotor support included body pitch and roll motion. The basic design philosophy required a soft flapping stiffness and the use of elastomeric damping for lag stability. The through-hub flexbeams had a flap flap flexure inboard, then a torsionally soft cruciform section outboard. The external cuff or torque tube was shear-restrained at the inboard end, to minimize the couplings and flexure loads due to pitch link shear forces. An elastomer was used at the cuff restraint to augment the inplane structural damping. The damper-restraint was oriented 11° nose down, so that with the trailing edge pitch link a negative pitch/lag coupling (stabilizing) was produced for collective angles above 11°. The rotor had a flap frequency of 1.04/rev, lag frequency of 0.69/rev, and 2.75° of precone. Hover stability (eigenvalues) was calculated using a modal analysis (the modes included the effects of the redunant load path) and dynamic inflow. Forward flight stability was calculated from time histories, using the C81 program. The model rotor tests showed an instability at the body roll resonance (at 0.675/rev), but full scale flight tests showed significantly higher damping, with no ground resonance problem. The model was gimballed at a point corresponding to the aircraft center of gravity (for air resonance simulation), while for the aircraft on the ground the rotation point was below the landing gear. The lower roll moment of inertia in the former case was sufficient to introduce the instability. Most of the parameters investigated experimentally showed little effect on the stability of this rotor; the built-in lag damper provided sufficient stability. The influence of droop and sweep were predicted well for the isolated rotor case (Fig. 114). While droop and sweep produced measurable damping changes for the isolated rotor, their influence was negligible for the coupled rotor/body case (the hover analysis still predicted an unfavorable influence of sweep, however). Forward flight increased the damping at the body roll mode resonance (70% rotor speed), but had little influence at the pitch mode resonance (Fig. 115). The analyses were accurate for the baseline configuration. Trends with some parameters (precone, damping, body motion) were predicted well, while others (sweep, control system stiffness) were not. The forward flight predictions were generally less accurate than the hover predictions.

4.2.5 Experimental Main Rotors

Seitz and Singer (Ref. 120) and Kloppel, Kampa, and Isselhorst (Ref. 102) described an experimental bearingless main rotor. The analysis used to design the rotor was a rigid blade and hinge spring model. A key parameter was the blade-to-beam droop. With low flap stiffness, the flap and lag bending take place inboard of the blade pitch change. Hence, the effective pitch/lag coupling depends primarily on the droop

$$K_{P_\zeta} = K_\zeta \beta_d / K_c$$

so negative droop will provide the desired (stabilizing) negative pitch/lag coupling. The experimental main rotor used BO-105 blades and hub. A T-beam was used (for small control force to twist the blade, and low stress) with a midchord torque rod. A damper, consisting of an elastomeric layer bonded to the flexbeam and covered by a stiff carbon fiber beam, provided an increase in lag structural damping of about 0.5% critical (50% higher; see Ref. 98). The rotor had a flap frequency of 1.10/rev and lag frequency of 0.68/rev (compared to 1.12 and 0.67 for the BO-105 hingeless rotor), -2° of droop and 1° of precone. Whirl tests showed that the rotor had less damping than the BO-105, even with the lag damper (Fig. 116). Ground resonance calculations indicated that stiffening of the BO-105 gear would be required to move the body pitch mode resonance from 104% rpm to 108% rpm. Air resonance calculations showed no problem, although the stability would be less than that of the BO-105. The reduced flap stiffness helped by lowering the roll mode frequency; however, a design with higher damping level would be preferred.

Seitz and Singer (Ref. 120) described two bearingless main rotor designs. The first design used a single flexbeam and torque tube configuration. The flexbeam had a cruciform section for torsion, with a flat flexure at the root for low flap frequency (1.07/rev). The elliptical, outer torque tube was constrained in shear by a snubber at the root (raising the lag frequency to 0.70/rev). The second design used a double flexbeam and mid torque rod configuration. The flexbeams had a T-section for torsion, and a flat flexure at the root (flap frequency 1.07/rev). Twin beam behavior was observed in the second lag mode in particular. Composite materials were essential to achieve the required tailoring of flexbeam properties.

4.2.6 ITR

Bousman, Ormiston, and Mirick (Ref. 111) discussed the bearingless hub design trends evident in the results of the U.S. Army/NASA Integrated Technology Rotor (ITR) program. The hub design goals included: hub drag D/q = 0.15% rotor disk area (performance); weight = 2.5% gross weight (performance and cost); parts count = 50 (cost and maintenance); hub moment stiffness = 1.03/rev flap frequency (vibration, gust response, handling qualities); hub tilt capability without fatigue = 5°, fatigue life = 10,000 hr, mean time between removal = 3000 hr (reliability and maintainability); provision for lag dampers; torsion stiffness such that swashplate actuator loads = current levels; and low production costs. Each of these goals could be achieved separately, but it was difficult to obtain all of them at once. It was decided to relax the flap frequency goal to 1.05/rev and the hub tilt goal to 4° for the next phase. The low flap frequency goal led to some consideration of gimballed or flap-hinge designs (with lag and pitch flexbeams), but most of the configurations examined were bearingless, and all were soft-inplane designs. The flexbeam design considerations were strength and fatigue life, with the low hub moment stiffness. The flexbeam could have a cross section varying along its length (with a flap flexure inboard and torsion section outboard), or not (which would be simpler to make and would avoid structural problems at the section transitions). Single, twin, and quadruple beam configurations were examined; a laminated beam was

also considered, for lower flap stiffness. Sometimes a shoe was required to control the flap bending curvature (on low stiffness designs). Many cross sections were possible for the torsion section of the flexbeam. The pitch control design considerations were weight, drag, and aeroelastic couplings. All the designs considered in these investigations included a shear restraint to react the control load at the root, so the control introduced was a pure torque. Without such a shear restraint, control input would produce a flap deflection also, hence more pitch link travel would be required; and the effective pitch/flap and pitch/lag couplings would be more complicated. The pitch control options included having the torque structure separate from the flexbeam or enclosing it; having the torque structure carry bending loads or not; and perhaps using an elastomeric damper in the shear restraint (which required that the torque structure be stiff in chord bending, or offset chordwise). Probably 1% to 3% damping could be obtained from structural damping, which would likely not be sufficient for aeromechanical and aeroelastic stability. The use of elastomeric dampers could give 3% to 6% damping, which would be acceptable. The dampers could be combined with the shear restraint, or could be a constrained layer of elastomeric material on the flexbeam. Analytical tools could not yet provide the detailed guidance needed for selecting aeroelastic couplings, but negative droop for negative pitch/lag coupling and beam prepitch for flap/lag structural coupling were desirable. Regarding materials, composites were essential for the required strength and the ability to achieve the separation of bending and torsion stiffnesses. Graphite tended to give a lighter, more compact, lower flap-stiffness design (from good stiffness to weight ratio); while fiberglass had better fracture toughness and failure modes for reliability and maintainability.

4.3 Bearingless Rotors--Tail Rotors

Bearingless tail rotor designs have been developed with the same goals as for main rotors: simplicity, with the resulting reduced weight and cost; improved maintainability; and improved survivability. There are two major differences compared to main rotors: the loads penalty is less severe, so most tail rotor designs are stiff-inplane; and several bearingless tail rotor designs are either ready for or in production.

Maloney and Porterfield (Ref. 121) developed an experimental bearingless tail rotor for the UH-1H helicopter. The blades had dual, fiberglass flexbeams; extensions of the airfoil formed a torque tube, with a shear reaction bearing at the root. The rotor was designed as a teetering hub with 35° of pitch/flap coupling (standard UH-1 configuration). A flap-lag instability was encountered in whirl tests because the lag frequency (about 1.3/rev) was lower than expected. Locking out the teeter motion eliminated the instability, but the increased hub moment limited the envelope. Analyses were not particularly helpful for this problem.

Fenaughty and Noehren (Ref. 122) described the development of the bearingless tail rotor for the UH-60 helicopter. Extensions of the blade spars formed through-hub flexbeams of uniaxial graphite/epoxy. With graphite rather than fiberglass, a smaller cross section could be used, hence a lower weight and lower torsion stiffness were possible. Extensions of the blade skin formed an external torque tube of fiberglass. Originally the torque tube was restrained inboard by the control system only, which allowed pitch/bending coupling. A snubber was added to negate the coupling and eliminate lost motion. The flap frequency (about 1.25/rev) and lag frequency (1.6 to 1.7/rev) were kept separate for flap-lag stability. Offset of the zero lift axis (aerodynamic pitch) above the flexbeam structural axis kept the lag frequency above 1.6/rev over the entire collective range. Flight tests showed that flap-lag stability was no problem. It was estimated that the bearingless design reduced the weight by 30% and the number of parts by 25%.

Shaw and Edwards (Ref. 123) developed a bearingless tail rotor for the YUH-61A helicopter. They were particularly concerned about survivability, so chose to achieve stability through flap/lag coupling rather than by frequency separation. The through-hub flexbeam was a thin, wide strap of fiberglass. Fiberglass was chosen over boron or graphite for its survivability characteristics: less brittle and slower propagation of severe damage. The strap was wide for survivability and thin for low hub moments. The rotor was thus stiff-inplane, and with 65° of pitch/flap coupling the flap and lag frequency separation was small. A rigid pitch horn was used for collective control (no shear restraint). In wind tunnel tests, the rotor initially encountered flap-lag instbilities in cyclic and reactionless modes (which had slightly different lag frequencies; Fig. 117), and a stall excited flap-lag-torsion oscillation at the second flap/first torsion mode frequency (Fig. 118). The theoretical tools available included a modal frequency analysis, and a rigid blade and hinge spring stability analysis; these tools provided guidance but were not sufficient for predicting absolute levels of stability. The wind tunnel tests established several parameters with favorable influence on stability: sweep, which introduced flap/lag and aerodynamic coupling; tip weights, which changed the frequencies; and a blunter leading edge contour, which eliminated leading edge stall.

Huber, Frommlet, and Buchs (Ref. 124) described the development of a bearingless tail rotor for the BO-105 and BK-117 helicopters. Soft-inplane designs were considered to minimize oscillatory inplane loads, reduce weight, and reduce control loads. The helicopter airframe modes were such that ground and air resonance would be no problem (no airframe modes were within the 6 to 14 Hz range of the tail rotor regressing lag mode). Two rotors were designed. The first was a three-bladed rotor. The fiberglass flexbeam consisted of twin C-beams, converging at the blade, where they formed the blade spar. A rigid pitch arm (metal or composite) with no shear restraint would be used. The rotor would have a flap frequency of 1.03/rev, 45° of pitch/flap coupling, and a lag frequency of 0.65/rev. The second was a four-bladed rotor. The single element, through-hub, fiberglass flexbeam had a flat flexure inboard for low flap stiffness, and a cruciform torsion section outboard. Damping elements consisted of four viscoelastic sheets on chordwise arms of the flexbeam, bridged by carbon-fiber plates. A rigid pitch arm (composite)

with no shear restraint would be used. The rotor would have a flap frequency of 1.04/rev, 45° of pitch/flap coupling, and a lag frequency of 0.69/rev. Counter-weights at the blade root would minimize control forces. These rotors were analyzed using the rigid blade and hinge spring model developed for the BO-105 rotor. Calculations indicated no problems with air or ground resonance: at least 1.5% inplane structural damping was needed, but 3.5% should be possible using the damping elements. These designs achieved a 20% weight reduction, and a 20% production cost reduction was predicted.

Blachere and D'Ambra (Ref. 125) described a Triflex tail rotor design. The rotor used a single arm flexbeam and a rigid pitch control sleeve, constrained at the root by a bearing. The flexbeam consisted of a bundle of roving threads (R-glass and epoxy resin) embedded in an elastomeric matrix for damping. The rotor had a flap frequency of 1.06 to 1.10/rev and a lag frequency above 0.5/rev.

Banerjee, Johnston, and Messinger (Ref. 126) described the development of an experimental bearingless tail rotor for the AH-64 helicopter. The through-hub, flat flexbeam formed an extension of the blade spar. The flexbeam was attached to the hub by elastomeric shear pads, such that the cyclic lag frequency was stiff-inplane (about 1.3/rev) for stability and low 1/rev loads; and the reactionless lag frequency was soft-inplane (about 0.7/rev) for low 2/rev loads and for damping from the elastomers. An external torque tube was used, with an elastomeric shear restraint on the inboard end. The flap frequency was about 1.2/rev. Negative pitch/lag coupling and 35° of pitch/flap coupling were obtained through pitch horn and pitch link geometry. Wind tunnel tests showed no stability problems over the operating range of the tail rotor.

4.4 Rotors with Circulation Control

In a rotor utilizing circulation control, a thin jet of air is blown from a spanwise slot along a rounded trailing edge. The jet remains attached over the curved surface due to the Coanda effect. Such blowing delays separation by energizing the boundary layer, and controls circulation by shifting the stagnation point. Hence with such a rotor, lift is controlled by the blowing as well as by the geometric pitch of the blade.

The lift coefficient is now a function of both angle of attack α and blowing coefficient C_μ = (jet momentum)/$\rho(1/2)V^2c$ (here ρ is the air density, V the airfoil section velocity, and c the chord). Then the perturbation lift force due to blade motion is:

$$\begin{aligned}\delta L &= \delta[(1/2)\rho V^2 c c_\ell] \\ &= (1/2)\rho V^2 c[c_{\ell_\alpha}\delta\alpha + c_{\ell_\mu}\delta C_\mu + c_\ell 2\delta V/V] \\ &= (1/2)\rho V^2 c[c_{\ell_\alpha}\delta\alpha + 2(c_\ell - C_\mu c_{\ell_\mu})\delta V/V]\end{aligned}$$

where $\delta C_\mu = -2C_\mu \delta V/V$ follows assuming that the jet momentum is constant during the motion. Normally an increase in the velocity (δV) implies a larger lift because of both the increased dynamic pressure and the decreased induced angle of attack. With a circulation control airfoil, the velocity increase in addition decreases the blowing coefficient, thereby decreasing the lift. A moderate amount of blowing will reduce the net lift perturbation due to inplane velocity perturbation; a large amount of blowing will change the sign of the lift perturbation. Trailing edge blowing may be expected therefore to alter the dynamic characteristics of the rotor. A moderate level of blowing will reduce the aerodynamic flap/lag coupling, and a large amount will change the sign. Hence there will be a tendency for flap-lag motion to be stabilized at low values of C_μ, while new instability regions (at low thrust with either high or low lag frequency) appear for high C_μ.

Chopra and Johnson (Ref. 127) analyzed the hover stability of rotors utilizing circulation control. The rigid blade with hinge spring model was used for flap-lag and flap-lag-torsion stability calculations. The general character of the dynamics was like that of conventional rotors. Instabilities were possible, but were mild and a low level of structural damping or flap/lag structural coupling would eliminate them (Fig. 119). An exception was the flap-lag instability of a soft-inplane rotor at high C_μ; structural damping or flap/lag coupling was not sufficient for stability. Pitch/lag coupling had a large effect on the stability, pitch/flap coupling less effect. A quasistatic torsion model was satisfactory for the pitch frequencies typical of rotors using circulation control. It was noted, however, that the aerodynamic pitch damping was low for the pitch axis at the mid-chord (a possible design choice with such rotors), so adequate torsion structural damping was important. Calculations for configurations representative of the Kaman Circulation Control Rotor (Fig. 120; flap frequency 1.1/rev, lag frequency 1.4/rev, no flap/lag coupling, large pitch/lag coupling) and the Lockheed X-Wing (Fig. 121; flap frequency 1.8/rev, lag frequency 3.6/rev, full flap/lag coupling, moderate structural damping) showed no stability problems. Both rotors were tested in the wind tunnel at full scale, with no indications of instabilities.

Chopra (Ref. 45) used a finite element analysis to calculate the hover stability of a bearingless rotor with circulation control. A configuration with both leading-edge and trailing-edge pitch links on an external torque tube was considered. The structural damping and the damping at the torque tube shear restraint were neglected. There was a significant influence of the blowing level on the predicted stability (Fig. 122). High blowing coefficient (hence, low collective for a given thrust) was destabilizing at

low and moderate thrust. The instability was not particularly severe, so it should be possible to design the torque tube shear restraint with sufficient damping to stabilize the motion.

4.5 Rotor/Engine Dynamics

Much of the U.S.A. industry experience with rotor/engine dynamics problems has been reported in efforts sponsored by the U.S. Army (Refs. 128-132). The problems and solutions have been categorized by Warmbrodt and Hull (Ref. 133). There are numerous cases of excessive rotor-induced vibration of the propulsion system; the usual correction involves modifying the structure (such as the engine mount) or weight to move the natural frequency away from the forcing frequency. A second type of problem is excessive vibration (forced or self-excited) because of engine/drive-train/rotor resonances; this has been corrected by modifying the rotor dynamic characteristics. A third type of problem is engine/drive-train torque oscillations, often involving a high gain fuel control system; this problem may require modifications to the drive-train flexibility, the fuel controller, or the blade lag dampers. A fourth type of problem is excessive main rotor overspeed or droop during maneuvers, which is corrected by revising the engine/fuel control system.

It is a characteristic of the analytical tools available for these problems that either the propulsion system model lacks the detail of the rotor model, or conversely. The problems are multidisciplinary, but the analyses are not. High gain control systems are making it important to have good dynamic models of the propulsion system, but such models are usually either not available or not coupled with the good rotor models. The engine vibration problems usually involve complicated structural dynamics, that cannot be predicted well even with a finite element model of the airframe.

Fredrickson, Rumford, and Stephenson (Ref. 134) described a rotor speed governor problem that occurred on the CH-47C helicopter. In flight tests of a growth version of the rotor and engine, a 4.1 Hz oscillation in the engine shaft torque and rotor speed was encountered. The phenomenon was present in hover and on the ground, but not in forward flight. The oscillations were 8% to 10% of the maximum steady torque and fuel flow; the lag damper force oscillated below the preload value. The effect could be duplicated analytically only by stiffening the lag damper The mode involved consisted of the rotor-lag motion opposed by the transmission and engine turbine. This mode was predicted to be at 3.1 Hz rather than the observed 4.1 Hz, a difference attributed to the lag damper. By improving the math model of the lag damper, it was possible to calculate the oscillation well. The problem could be corrected by softening the lag damper, but that led to unacceptable ground resonance characteristics. Reduced fuel-control gain also worked, but was marginal in cold air. The final solution was to both reduce the gain and increase the time constant in the fuel control, such that the gain at 4 Hz was reduced by a factor of three (Fig. 123). The engine response to power demands by the pilot was not perceptibly degraded.

Fredrickson (Ref. 135) described a rotor/drive system 4/rev torsional resonance that occurred in the Model 347 tandem helicopter. High 4/rev blade chord bending moments were encountered in transition and in high speed at high gross weight. The mode involved consisted of the collective lag motions of the two rotors opposing each other through the shaft. A blade chord frequency at 5.3/rev produced a coupled blade and drive system frequency at 4/rev. The problem was corrected by raising the chord frequency above 6/rev, and hence, the coupled system frequency to 4.3/rev, by use of boron fiber doublers bonded to the blade trailing edge and boron skins applied to several blade boxes (the simplest solution, if not necessarily the best).

Twomey and Ham (Ref. 130) described two problems encountered on the CH-53E helicopter. The first problem was an oscillation of the rotor and drive system in the third torsional mode. In flight tests, 3.6/rev cockpit vibration occurred in specific flight conditions, with a time to double amplitude of 10 to 12 sec. The mode involved consisted of collective edgewise bending and the drive system torsion. The rotor blade motion was a combination of rigid and first elastic bending, such that there was little motion at the lag damper. An analysis including the fuel controller and the blade edgewise motion did not indicate an instability. It was speculated that the instability arose from coupling in forward flight of the 3.6/rev collective edgewise mode with a 2.6/rev cyclic flapwise mode. The cure involved reducing the blade edgewise stiffness (lowering the natural frequency to 3.45/rev to decouple it from the flapwise mode, and to increase the modal motion at the lag damper). For the flight tests the blade was softened by removing graphite strips that had been added to the trailing edge to improve the stress levels; the blades were redesigned for production. The second problem was a feedback oscillation of the rotor and drive system first torsional mode. In flight tests, a low frequency (2 Hz) oscillation of the aircraft occurred in forward flight. The mode involved consisted of the collective rotor lag motion, opposing the drive train and engine torsion, and the fuel controller. The background 1/rev motion in forward flight decreased the effectiveness of the lag dampers. A bench test was conducted to determine the equivalent viscous damping of the lag damper with a background of 1/rev and higher harmonic motion. Flight-test data on the harmonics of the lag motion then allowed a specification of the equivalent damping available as a function of flight speed and rotor speed. A good prediction of the stability was achieved when the reduction of the lag damping in forward flight had thereby been accounted for. The cure involved increasing the power turbine governor time constant from 0.165 to 0.7 sec, thus reducing the fuel controller gain by a factor of 15 at 2 Hz. This modification had little influence on the engine power response to pilot commands.

Thibert and Maquin (Ref. 136) discussed a transmission oscillation that occurred during development of a larger fan-in-fin tail rotor for the SA-365-N1 helicopter. A substantial torque oscillation at 4.4 to 5.8 Hz was observed in high-speed flight with slip, upon a sudden increase of the tail rotor pitch.

Analysis (a combination of linearized eigenvalue and nonlinear time history calculations) showed coupled main rotor and tail rotor modes at 4.6 and 4.9 Hz, involving little engine response; the 4.6 Hz mode was lower damped. Analysis and flight test showed that increased lag damping and increased governor time constant had little influence on the phenomenon. The analysis suggested that stiffening the tail rotor transmission by 33% would increase the frequency of the mode and so also increase its damping (by increasing the coupling with the 4.9 Hz mode, which then became less damped). The correction, confirmed by flight tests, involved replacing a duralumin central tail transmission shaft with a steel shaft. For the production aircraft it was possible to simply thicken the shaft (a lighter weight solution).

4.6 Articulated Rotors

It should not be assumed that articulated rotors are without interesting dynamic phenomena. Many problems encountered in the development of more conventional design concepts simply are not reported. A couple of examples will serve as notice against complacency.

Silverthorn (Ref. 137) described an advancing whirl mode instability encountered on an articulated main rotor. The phenomenon involved the rotor cyclic motions (rigid flap, lag, and pitch, with little bending), flexibility of the rotor support structure, and cyclic pitch/mast-bending coupling during pitch and roll motion of the hub relative to the fuselage. A 14 to 15 Hz (about 3/rev) instability was predicted to occur at 104% normal rotor speed (Fig. 124). The rotor was predicted to be stable without the influence of aerodynamics or the pitch/mast-bending coupling. In whirl tests the instability was encountered at 119% rpm, still below the required stability margin of 120% rpm. Better correlation with theory was achieved using measured structural damping and eliminating a three-dimensional aerodynamic-center shift at the blade tips. The analysis suggested that a forward shift of the blade center-of-gravity would help, but that implied a blade redesign and weight increase. The cure adopted involved increasing the support structure stiffness, so the stability boundary was well above 126% rpm in both whirl and flight tests. Adding swept tips (hence moving the aerodynamic center aft relative to the center-of-gravity) also stabilized the motion.

Neff (Ref. 138) described an instability encountered in an experimental articulated rotor on the OH-6A helicopter, involving the first elastic chord, the second elastic flap, and the first reactionless torsion blade modes. Shortly after entry to autorotation and establishment of stabilized descent, the main rotor blades abruptly went out of track and a severe vibration was felt in the controls and the airframe. The pilot applied power before the static load limits were exceeded. The blade loads data indicated a 4.63/rev mode with -0.6% damping. Analysis had initially predicted stability, with a first chord frequency at 4.4/rev. A fixed root was introduced for the chord bending boundary condition (with a four-stage friction lag damper, the blade was probably fully restrained at the mean position of autorotation), and effective pitch/lag and flap/lag coupling due to the mean out-of-plane bending was added (the flap bending at the tip significantly increased in autorotation). The analysis still predicted stability, but the first chord frequency was at 4.75/rev. Finally, the reactionless control system stiffness rather than the cyclic stiffness was used (an increase by a factor of seven, from 13,600 to 100,000 in.-lb/rad). Then the mode was predicted to be unstable, at 4.66/rev and -0.65% damping (Fig. 125). The theory suggested that an aft shift of the tip weight (with the net blade center-of-gravity still forward of the aerodynamic-center) would stabilize the motion, which was confirmed by flight tests. Subsequently, a similar problem was predicted to occur in a growth version of the production rotor. An aft shift of the tip center-of-gravity was introduced, and there were no stability problems in the flight tests.

4.6 Tilting Proprotor Aircraft

Investigations of the dynamics of tilting proprotor aircraft have generally focused on the whirl-flutter stability. Whirl flutter is a coupled motion of the proprotor and the airframe (typically the wing elastic modes) that becomes unstable at high forward speed. The rigid body and elastic motion of the blades makes tiltrotor whirl flutter a different, and more complicated, phenomenon than the whirl flutter of a propeller-driven airplane. Johnson (Ref. 139) assessed the present capability to predict tilting proprotor dynamics. Considerable work has been done and confidence gained on predicting whirl flutter stability. New designs will require the ability to analyze new hub configurations, and likely will require a better treatment of high-speed aerodynamic effects on the rotors. Most tiltrotor designs, including the gimballed rotor of the XV-15, have dynamic characteristics similar to those of hingeless rotors, notably the importance of pitch/lag and flap/lag coupling. In addition, the tiltrotor must operate over large ranges of rotor speed and collective pitch. Rotor loads remain important, since they can define the upper limit of the conversion corridor. Oscillatory loads on the airframe, particularly the nacelle and wing, can be a problem (normally cured by good structural design or structural modification, rather than by accurate prediction). Basically the tilt rotor configuration eliminates most concerns with fuselage vibration. Even in helicopter mode, the wing dynamics provide some vibration absorption, and the rotors can be tilted forward to minimize the wake-induced vibration at low speed.

Generally, the aerodynamic analysis is simpler for the proprotor (high inflow, axial flight) than for the helicopter rotor (low inflow, edgewise flight). Axial flight implies a symmetric aerodynamic environment, hence constant-coefficient equations of motion. In high inflow, both the inplane and the out-of-plant blade motion produce a first order change in the blade angle of attack, hence through the lift-curve slope a first order change in lift, which has both inplane and out-of-plane components. So the lift-curve slope terms dominate the aerodynamic forces (Ref. 140), which depend then mainly on the Lock number and the ratio of flight speed to tip speed. In contrast, for the rotor with low inflow, inplane motion produces lift and drag perturbations due to the dynamic pressure change, and tilts the mean lift and drag

forces; so the inplane forces or forces due to the inplane motion are small, and depend on the blade trim loading (see Ref. 7). Lift changes due to angle-of-attack perturbations, normally responsible for the high aerodynamic damping of the rotor flap motion, in the proprotor also produce a high aerodynamic damping of the blade inplane motion. Another result of high inflow is the large collective pitch and built-in twist required; and operating in both helicopter and airplane modes requires a large range of collective pitch.

For the gimballed, stiff-inplane proprotor design, blade pitch motion has a significant influence on whirl flutter (Fig. 126), through the introduction of effective pitch/lag coupling (Refs. 141 and 142). The blade precone is normally selected for hover, so in propeller configuration the precone is too large. There will be a downward elastic coning deflection of the blade. With no droop and small thrust, the effective pitch/lag coupling is negative and proportional to the precone. Negative pitch/lag coupling has a destabilizing influence on the whirl flutter. Figure 127 shows the stabilizing influence of reduced precone or increased control system stiffness, through the reduction in magnitude of pitch/lag coupling. Blade droop has a similar effect, while not increasing hover coning loads as does reduced precone (since droop becomes aft blade sweep at the low collective pitch angles of hover). The blade inplane motion has an effect on whirl flutter stability levels also (Fig. 126), particularly at resonances of the regressing lag mode with a wing mode (Ref. 140). With a soft-inplane rotor, air resonance is possible at low-flight speeds, particularly involving the wing vertical-bending mode (Fig. 128). At operating flight speeds, the air resonance is stabilized by the aerodynamic lag damping in high inflow and the wing aerodynamic damping.

With increasing Mach number, the blade lift-curve slope first increases, which increases the aerodynamic forces involved in whirl flutter, and so has an unfavorable influence on the stability (Fig. 129). After lift divergence (at a Mach number of around 0.7 to 0.8), the lift-curve slope decreases. If the blade section Mach number is above the lift divergence Mach number over a large fraction of the blade tip, the reduction in aerodynamic forces will significantly increase the stability. This phenomenon becomes particularly important as the speed of sound decreases at higher altitude (Ref. 142).

The rotor rotational-speed degree-of-freedom has a major influence on the whirl flutter stability (Refs. 140 and 143). Vertical bending of the wing is accompanied by a roll motion of the rotor shaft. If the rotor rotational speed is fixed relative to the pylon, this roll motion will be transmitted to the rotor, and the high aerodynamic damping of the rotor will greatly stabilize the wing mode (Fig. 130). If the rotor is windmilling, the rotational degree of freedom will be free relative to the pylon, and this source of damping will be absent. Typically, the engine inertia, engine damping, and rotor-speed governor offer little restraint of the rotational-speed degree of freedom in the symmetric motions of a tilting proprotor aircraft. The difference between powered and windmilling stability (Fig. 130) is primarily due to the difference in trimmed blade deflection. In the antisymmetric motions, however, the interconnect shaft constrains the rotor speed, introducing a differential speed mode with a natural frequency of the same order as the wing modes.

Tilting proprotor stability can be analyzed using a rigid blade and hinge spring model for the rotor. As for hingeless and bearingless helicopter rotors, the key to success with such theories is in the correct specification of the effective pitch/lag, pitch/flap, and flap/lag couplings. Elastic flap-lag-torsion rotor models have also been developed for tilting proprotors (Refs. 29 and 141). To the models developed for helicopter rotors, it is necessary to add high inflow aerodynamics, and the structural dynamics of blades with large collective and large twist. The calculations shown in Figs. 126-130 were produced using the analysis of Ref. 29.

Johnson (Ref. 144) presented a comparison of predicted and measured whirl flutter damping. Figure 131 shows the wing beam bending mode stability for a rotor windmilling on a cantilever wing in a wind tunnel. The calculations were also produced using the analysis of Ref. 29. The rotor was a small-scale model of an early gimballed hub design for the Bell/Boeing JVX aircraft.

Bilger, Marr, and Zahedi (Ref. 145) described the dynamic characteristics of the XV-15 Tilt Rotor Research Aircraft. Whirl flutter stability and blade loads were no problem in the aircraft flight tests. Initially the pylon loads (in the conversion spindle and downstop) were high (Fig. 132). A pylon lateral mode was excited at 2/rev; the source of the excitation was the second cyclic rotor mode loads at 1/rev and 3/rev, acting through the gimal. The correction involved reducing the downstop stiffness by a factor of 4.4, in order to move the load peak (at resonance with the pylon lateral mode) below the rotor speed of interest. Initially the loads in the engine coupling gearbox were high as well (Fig. 133). The 2/rev excitation forces were reduced by optimizing the lateral and longitudinal cyclic pitch in airplane configuration to maintain zero flapping at high speed. The reduction in downstop stiffness also reduced the 2/rev engine loads. These problems were both associated with the design of the gimbal, and the complicated load paths in the pylon.

5. CONCLUDING REMARKS

5.1 Basic Dynamic Problems

Stability concerns for advanced rotorcraft have centered on flap-lag stability and air/ground resonance. The pitch/lag coupling and structural flap/lag coupling have a major influence on the stability.

The effective coupling is a result of the rotor blade nonlinear dynamics, and depends on the detailed hub parameters.

Simple analytical models of hingeless rotors have been derived for research purposes and to support aircraft development. There has been experimental verification of these models, and they have provided much understanding of the basic dynamic phenomena. There have been strong research programs to develop elastic blade analyses for hingeless rotors. There has been limited experimental verification of the models, and they are starting to be used to support aircraft development. For bearingless rotors, simple theoretical models are less useful, because the dynamics of the actual configurations are so complex. As usual, designers of new rotors are ahead of the analyzers.

Rotor loads is the forced response problem, requiring the full nonlinear solution rather than just the linearized equations, and much more attention to aerodynamics. Generally there have been advances in the scope of loads prediction capability, but not in the accuracy. The prediction of mean and oscillatory loads is acceptable for design purposes, but detailed examination of correlations shows that the phenomena are still not completely understood.

Rotor-induced vibration adds the airframe structural dynamics to the problem. The attention in aircraft development is on vibration reduction, either passive or active, rather than on vibration prediction.

5.2 Advanced Topics in Dynamics

In higher harmonic-control research, the promise of the self-tuning regulator concept is beginning to be realized. The dynamic inflow models are a productive start for routine use of unsteady aerodynamics in rotor dynamics. Finite elements bring needed flexibility to rotor analyses, but the large number of degrees of freedom introduces major difficulties with complexity and computation time. The analyses being developed for composite rotors are necessary to realize the potential design flexibility of the materials.

5.3 Dynamics of Rotorcraft Configurations

There are soft-inplane hingeless main rotors in production. Lower flap frequencies are desired, to reduce the vibration and gust response and to minimize adverse handling qualities effects. Vibration more often than stability has been a problem in hingeless rotor development.

Experimental bearingless rotors have been developed, in further pursuit of design simplicity. The main rotor designs are soft-inplane, while the tail rotor designs are mostly stiff-inplane. Perhaps the most common configuration involves a flexbeam with an inboard flap flexure (for low flap frequency), plus an external torque tube with a snubber/damper at the root (for control of the pitch/bending coupling and augmentation of the structural damping). Stability has been a major concern, particularly air/ground resonance.

In rotors using circulation control, the trailing-edge blowing directly influences the lag dynamics, but the rotors tend to be very stiff. Coupled engine/rotor dynamic problems include local vibrations, and fuel controller dynamics with recent high-gain designs. Often the interdisciplinary nature of these engine/rotor problems is not fully reflected in the analyses. With articulated rotors, multimode dynamics can still provide surprises.

Concerns regarding tilting proprotor aircraft dynamics have focused on whirl flutter, which requires the addition of high inflow aerodynamics and high-pitch/high-twist structural dynamics to the analyses. The dynamic phenomena of hingeless rotors are generally a factor as well.

New rotorcraft configurations have generally been developed with the support of simple theories (or none); tests have been essential. Advanced analyses are only now beginning to help aircraft development. The designers remain one step ahead of the analyzers. A more flexible theoretical approach is needed, separating the helicopter and rotor configuration from the mathematical modeling. In tests of innovative designs, the real hardware may be expected to continue to provide interesting new dynamic problems.

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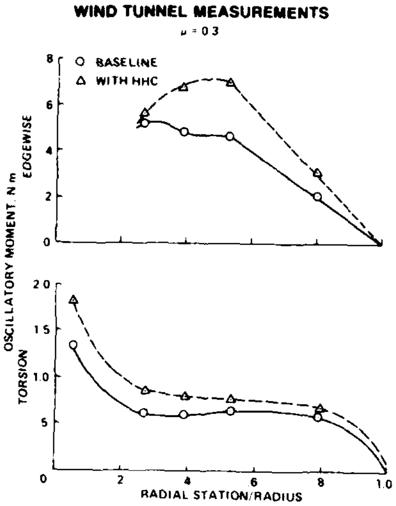

Fig. 78. Measured higher harmonic control of helicopter vibration (Ref. 76).

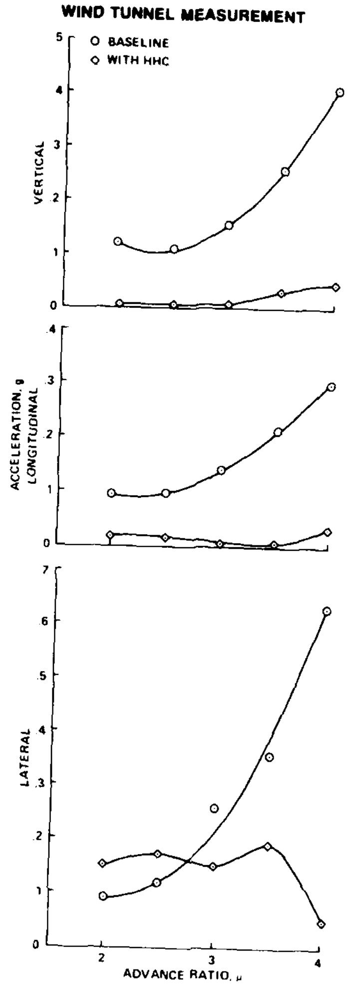

Fig. 79. Measured higher harmonic control of helicopter vibration (Ref. 77).

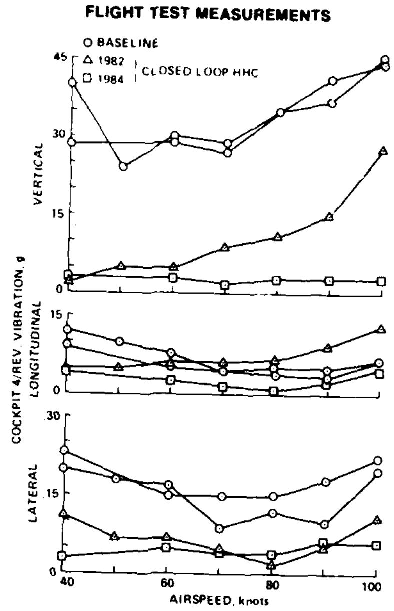

Fig. 80. Measured higher harmonic control of helicopter vibration (Refs. 78 and 79).

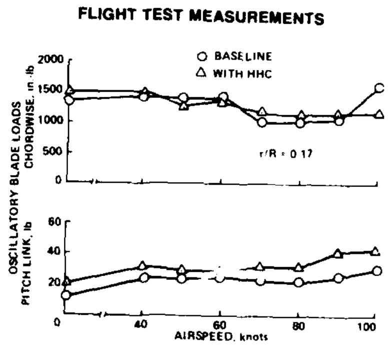

Fig. 81. Measured higher harmonic control of helicopter vibration (Ref. 78).

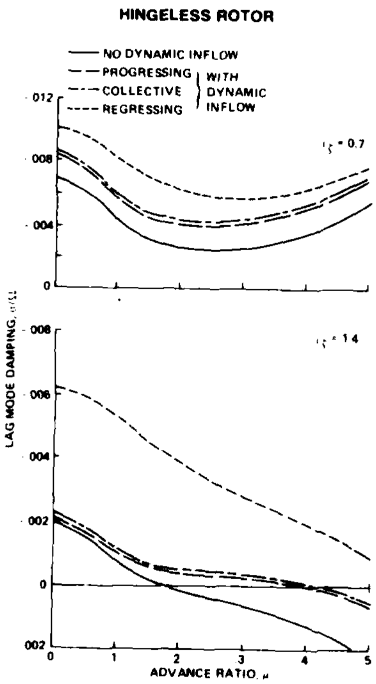

Fig. 82. Calculated flap-lag stability with dynamic inflow (Ref. 86).

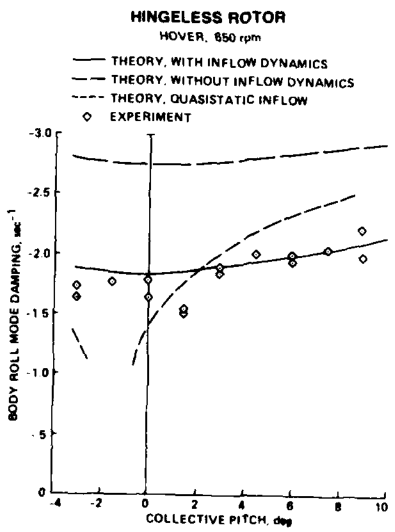

Fig. 83. Ground resonance stability with dynamic inflow (Ref. 80).

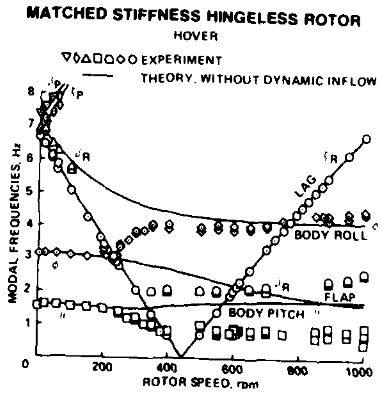

Fig. 84. Ground resonance stability with dynamic inflow (Ref. 80).

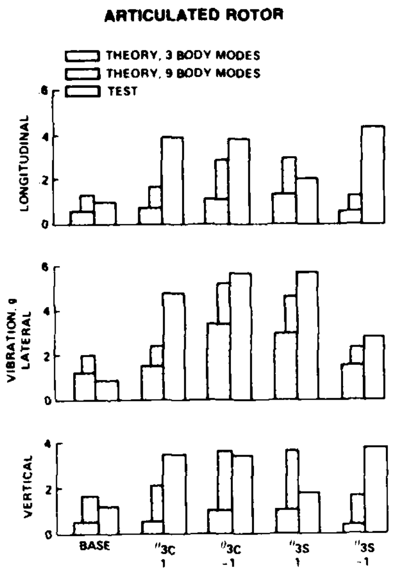

Fig. 72. Articulated rotor vibration (Ref. 63).

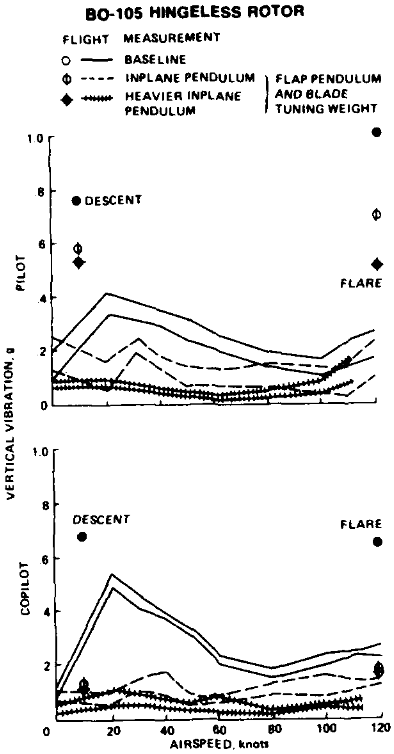

Fig. 73. Measured hingeless rotor vibration reduction (Ref. 68).

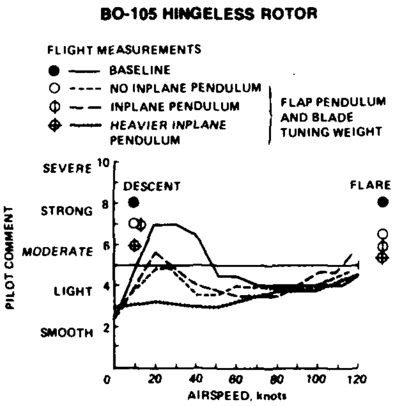

Fig. 74. Measured hingeless rotor vibration reduction (Ref. 68).

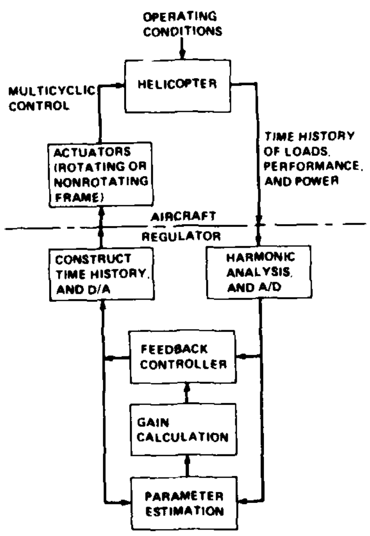

Fig. 75. Schematic of helicopter higher harmonic control system (Ref. 75).

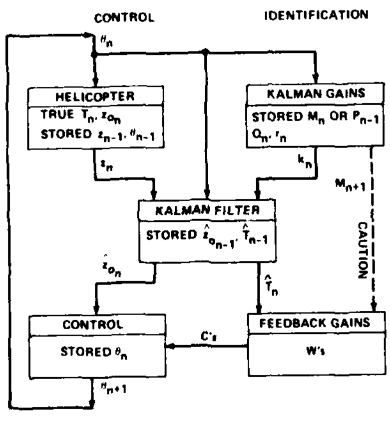

Fig. 76. Self-tuning regulator (adaptive open-loop version) for higher harmonic control of helicopter vibration (Ref. 75).

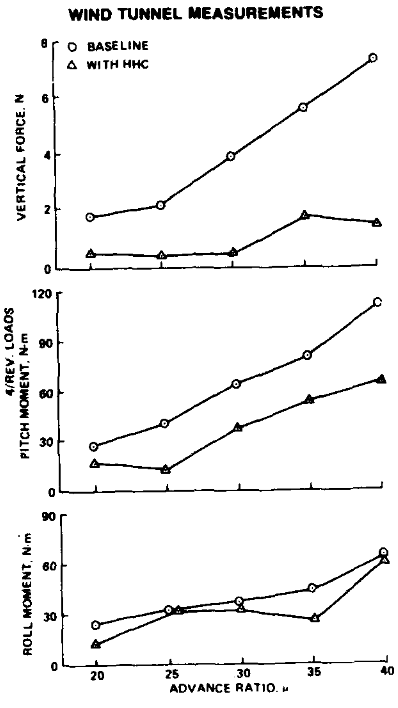

Fig. 77. Measured higher harmonic control of helicopter vibration (Ref. 76).

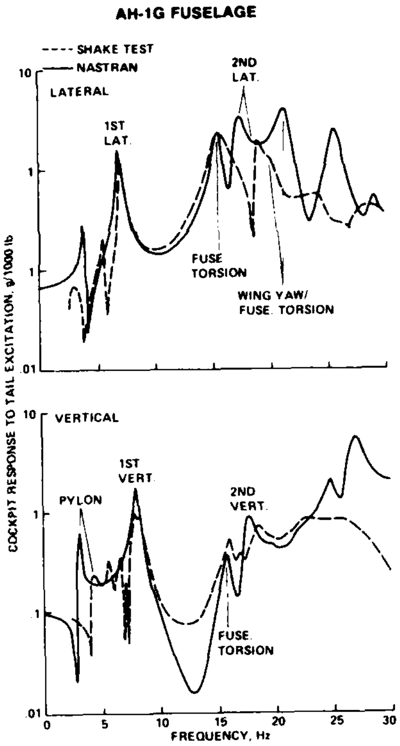

Fig. 66. Airframe structural dynamics (Ref. 65).

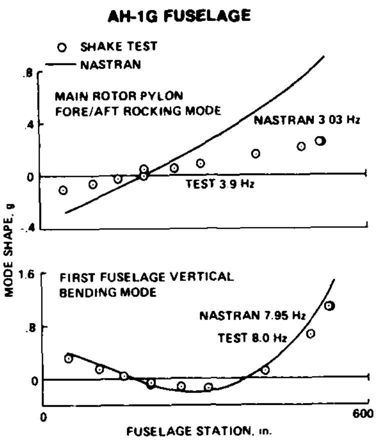

Fig. 67. Airframe structural dynamics (Ref. 65).

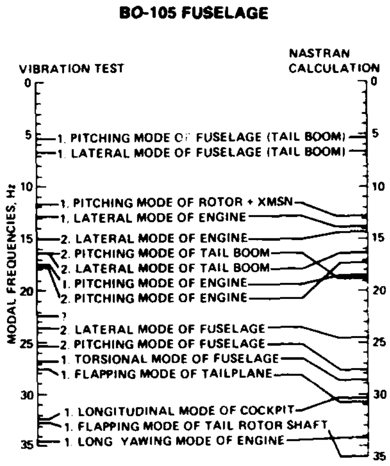

Fig. 68. Airframe structural dynamics (Ref. 66).

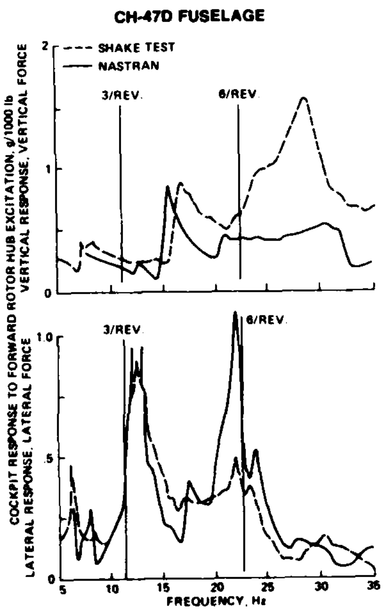

Fig. 69. Airframe structural dynamics (Ref. 67).

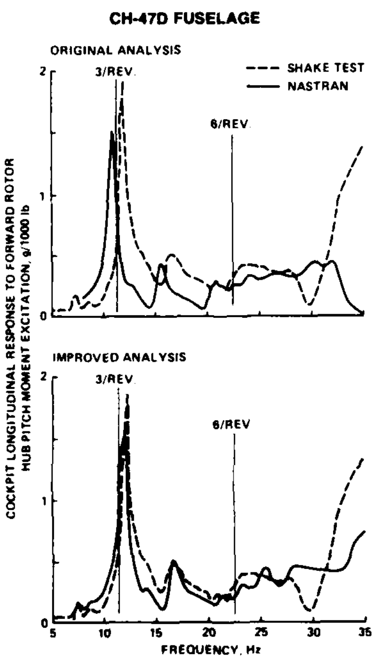

Fig. 70. Airframe structural dynamics (Ref. 67).

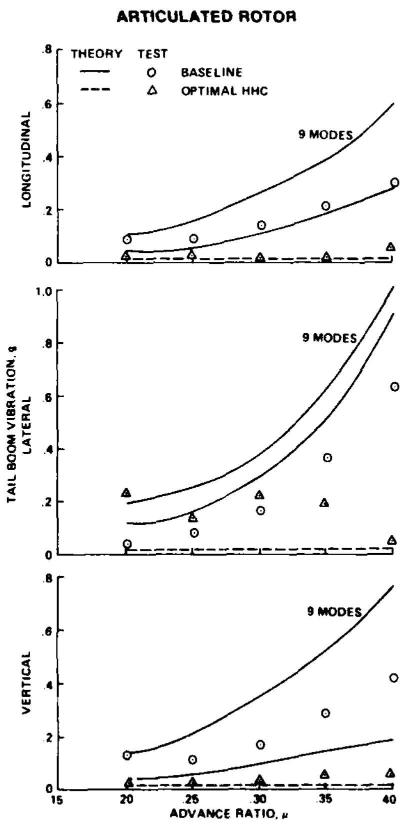

Fig. 71. Articulated rotor vibration (Ref. 63).

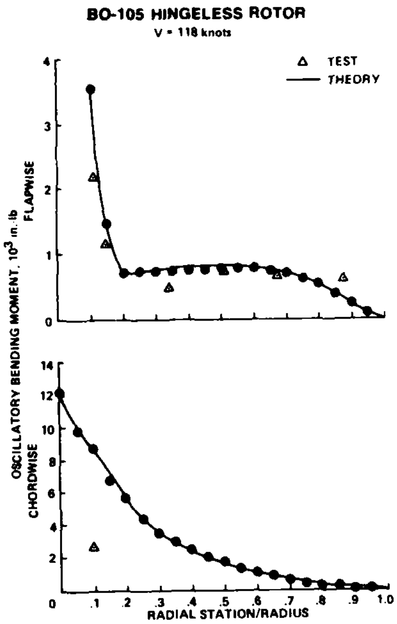

Fig. 59. Hingeless rotor blade loads (Ref. 59).

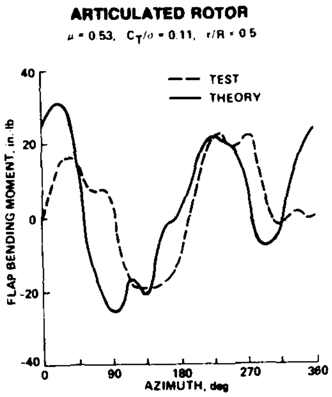

Fig. 60. Articulated rotor blade loads (Ref. 60).

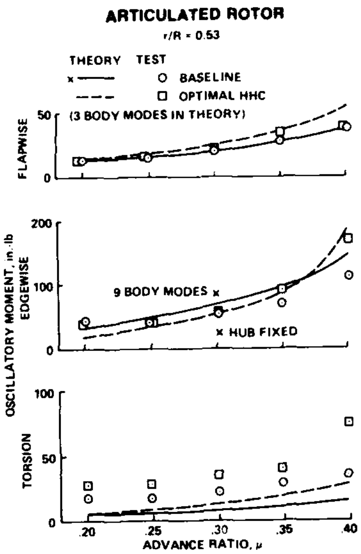

Fig. 61. Articulated rotor blade loads (Ref. 63).

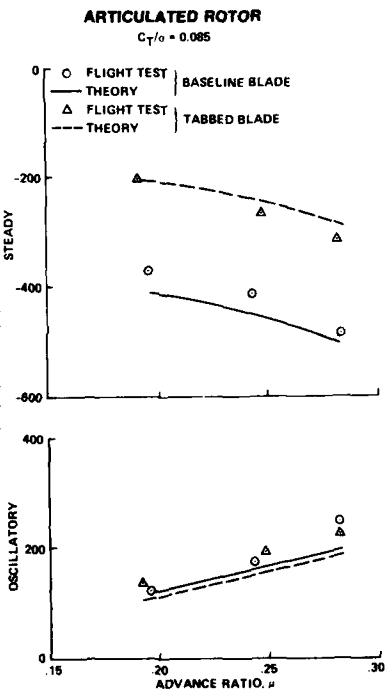

Fig. 63. Articulated rotor blade loads (Ref. 51).

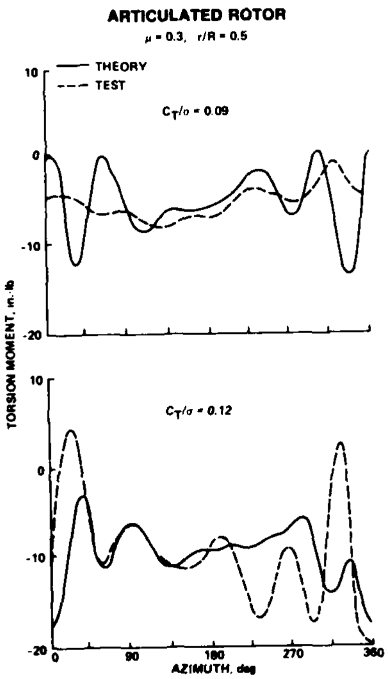

Fig. 62. Articulated rotor blade loads (Ref. 60).

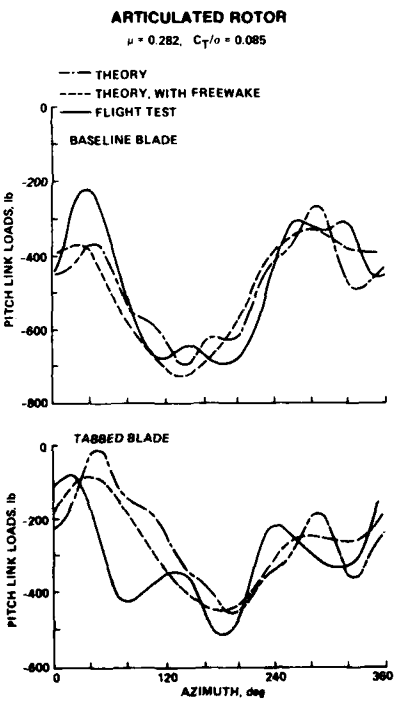

Fig. 64. Articulated rotor blade loads (Ref. 51).

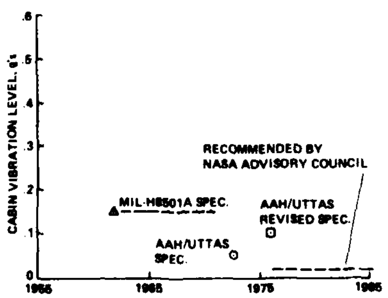

Fig. 65. Trend of helicopter vibration levels (Ref. 64).

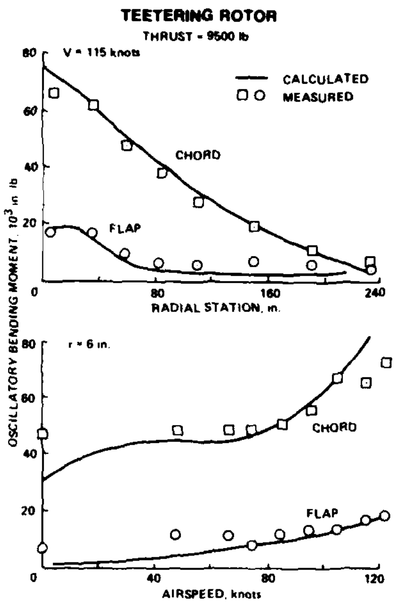

Fig. 53. Teetering rotor blade loads (Ref. 55).

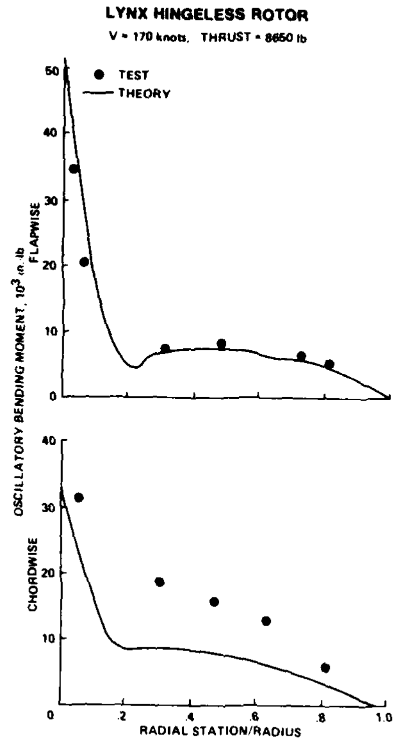

Fig. 54. Hingeless rotor blade loads (Ref. 56).

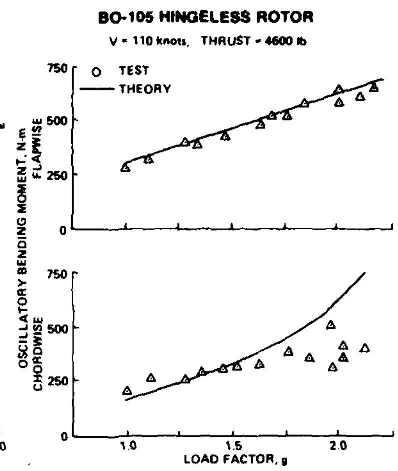

Fig. 55. Hingeless rotor blade loads (Ref. 57).

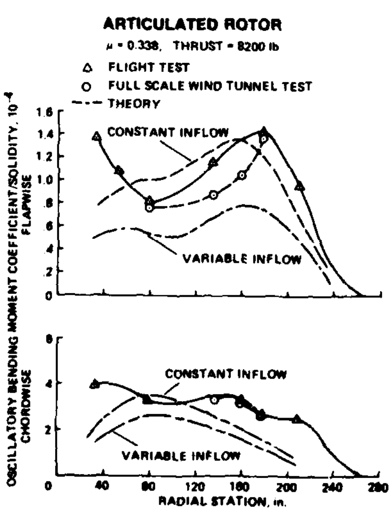

Fig. 56. Articulated rotor blade loads (Ref. 62).

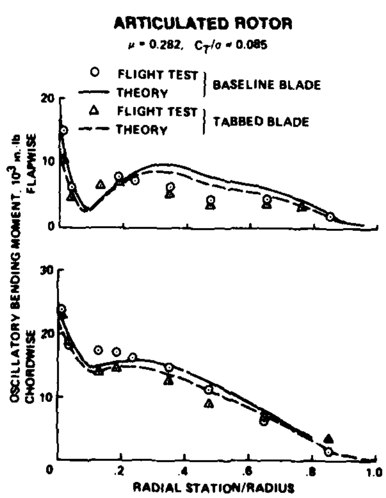

Fig. 57. Articulated rotor blade loads (Ref. 51).

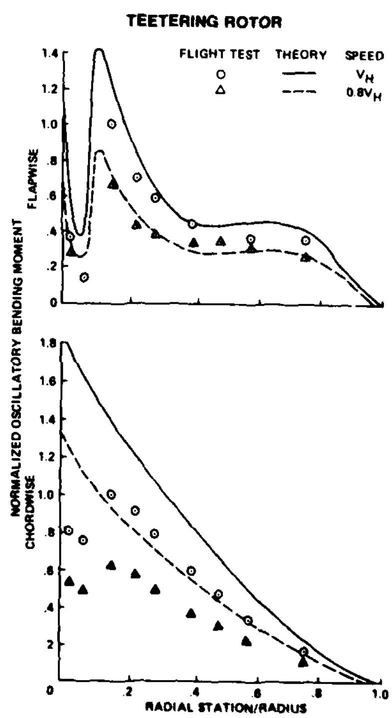

Fig. 58. Teetering rotor blade loads (Ref. 61).

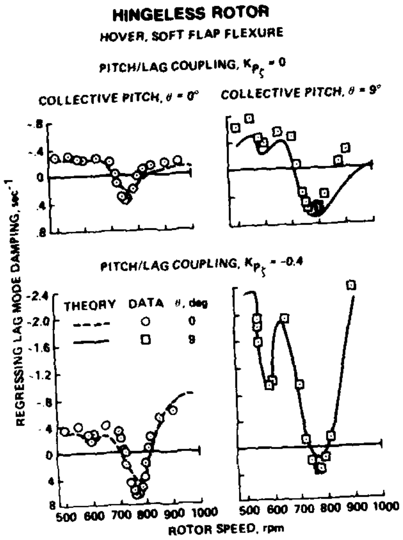

Fig. 47. Hingeless rotor ground resonance stability (Ref. 47).

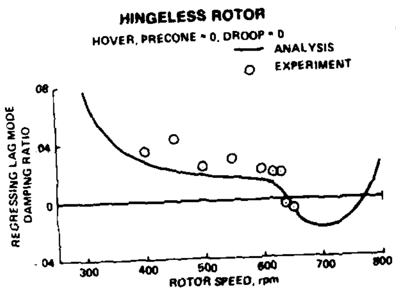

Fig. 48. Hingeless rotor ground resonance stability (Ref. 48).

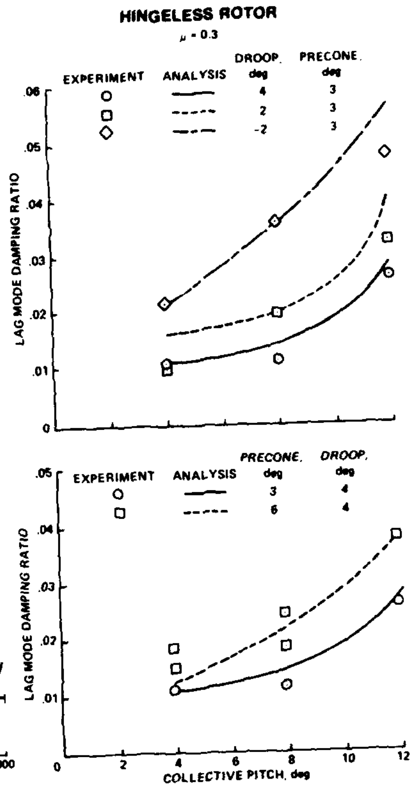

Fig. 49. Hingeless rotor ground resonance stability (Ref. 48).

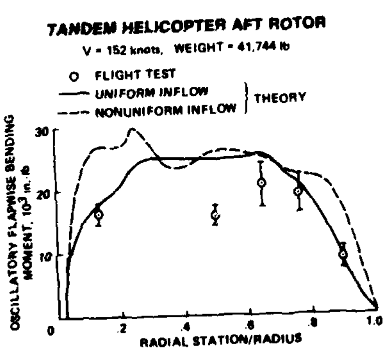

Fig. 50. Articulated rotor blade loads (Ref. 52).

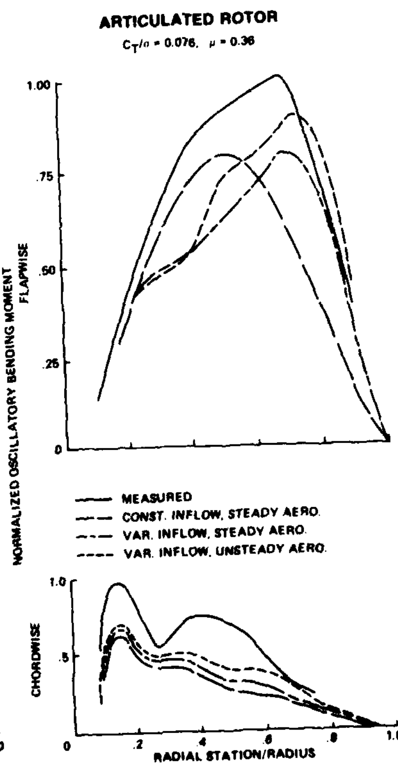

Fig. 51. Articulated rotor blade loads (Ref. 53).

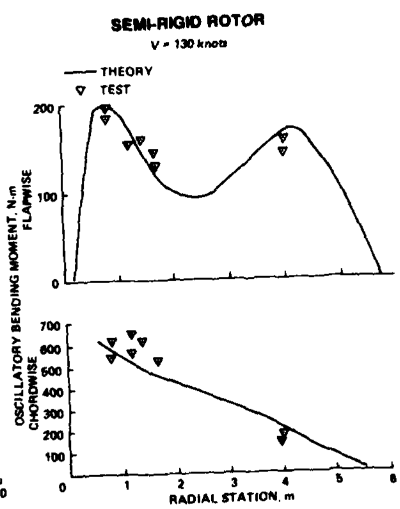

Fig. 52. Semi-rigid rotor blade loads (Ref. 54).

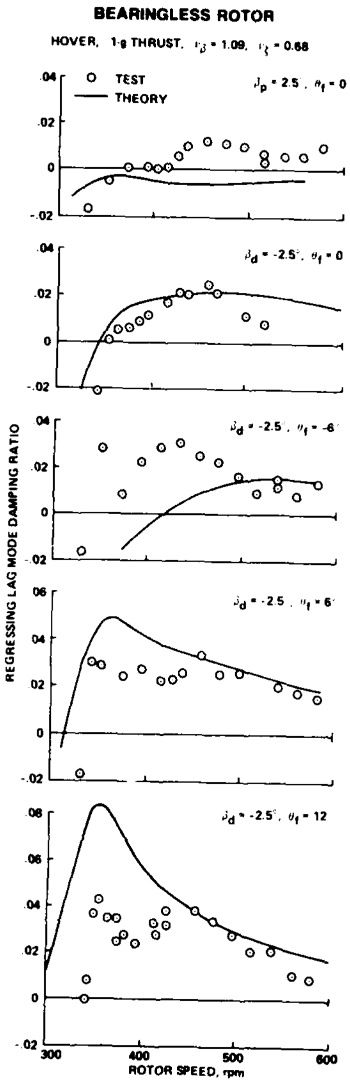

Fig. 39. Bearingless rotor stability (Ref. 39).

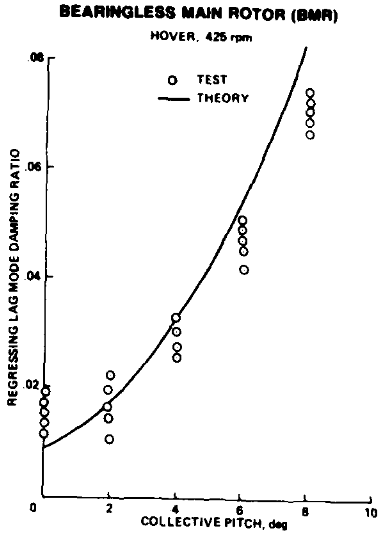

Fig. 40. Bearingless Main Rotor (BMR) stability (Ref. 40).

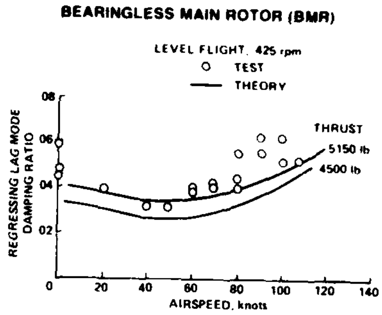

Fig. 41. Bearingless Main Rotor (BMR) stability (Ref. 40).

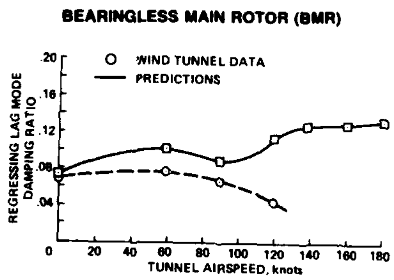

Fig. 42. Bearingless Main Rotor (BMR) stability (Ref. 41).

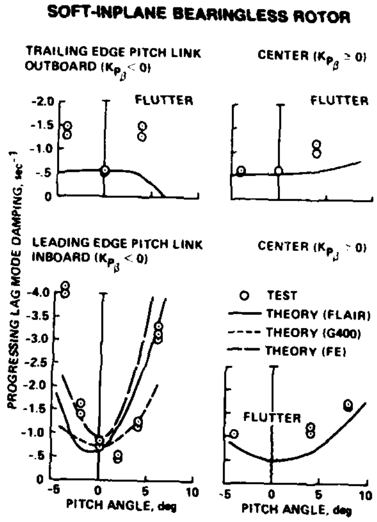

Fig. 43. Bearingless rotor stability (Ref. 42).

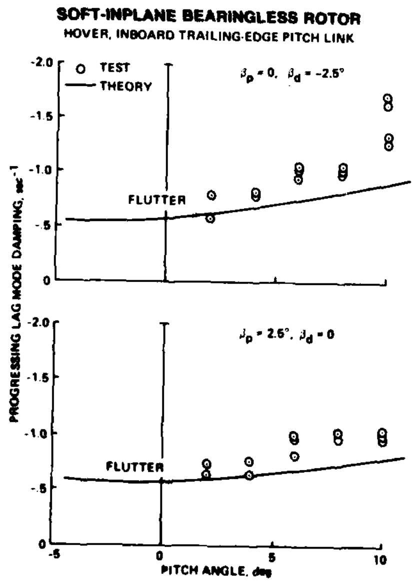

Fig. 44. Bearingless rotor stability (Ref. 42).

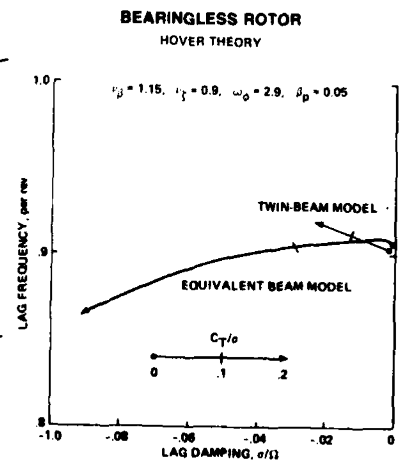

Fig. 45. Calculated bearingless rotor root locus (Ref. 44).

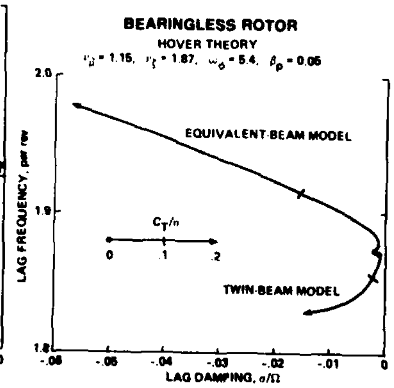

Fig. 46. Calculated bearingless rotor root locus (Ref. 44).

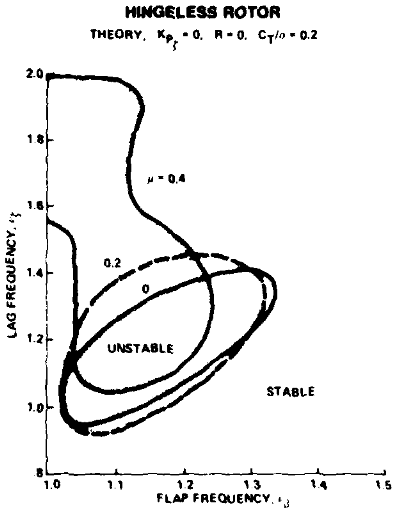

Fig. 33. Calculated flap-lag stability boundaries (Ref. 31).

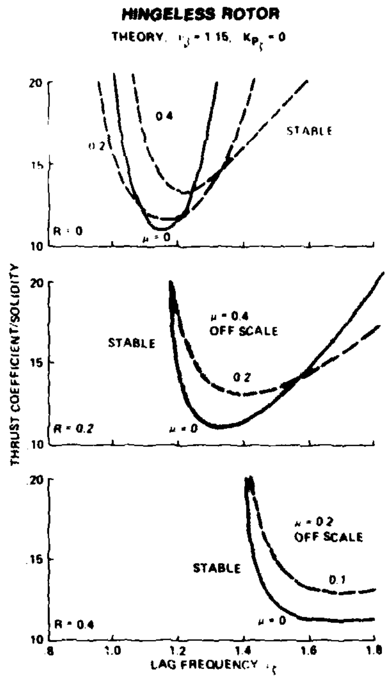

Fig. 34. Calculated flap-lag stability boundaries (Ref. 31).

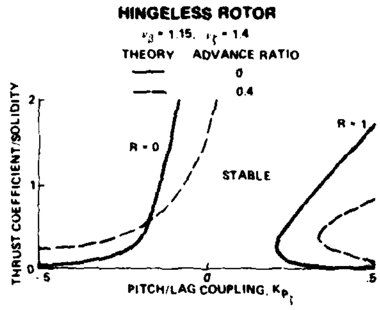

Fig. 35. Calculated flap-lag stability boundaries (Ref. 31).

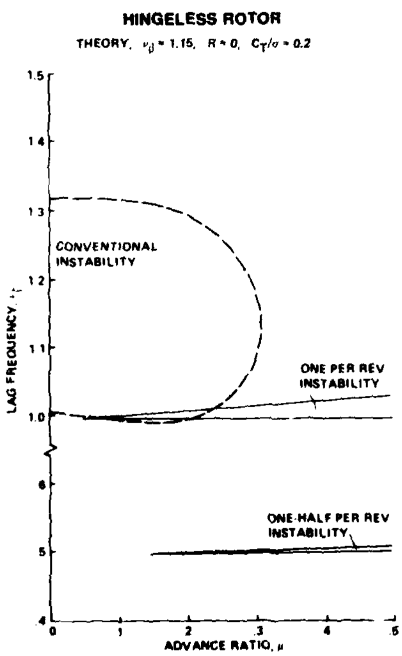

Fig. 36. Calculated flap-lag stability boundaries (Ref. 31).

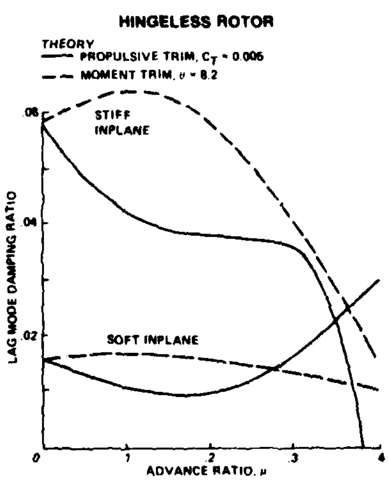

Fig. 37. Calculated elastic flap-lag-torsion stability (Ref. 34).

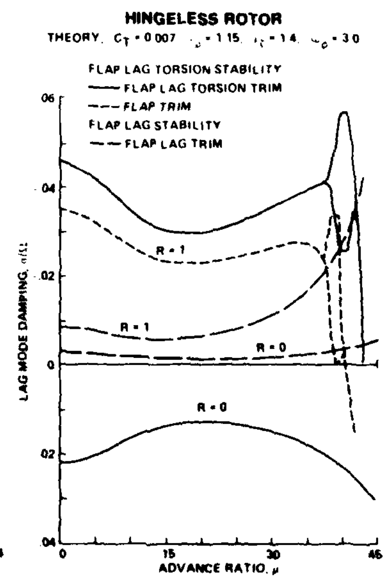

Fig. 38. Calculated elastic flap-lag-torsion stability (Ref. 35).

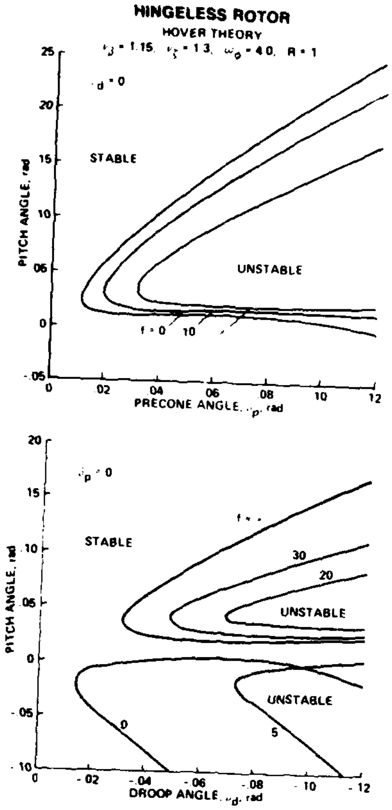

Fig. 25. Calculated pitch-flap-lag-torsion stability boundaries (Ref. 26).

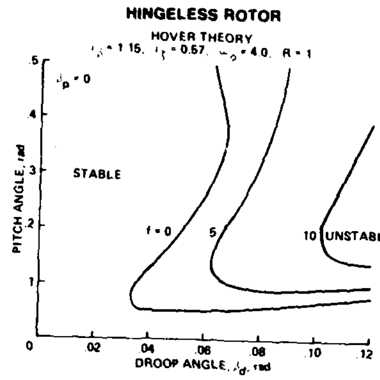

Fig. 26. Calculated pitch-flap-lag-torsion stability boundaries (Ref. 26).

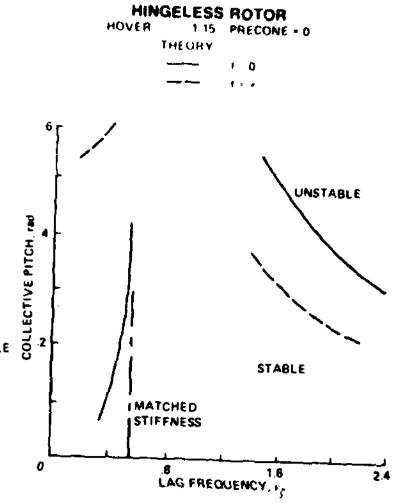

Fig. 27. Calculated pitch-flap-lag-torsion stability boundaries (Ref. 27).

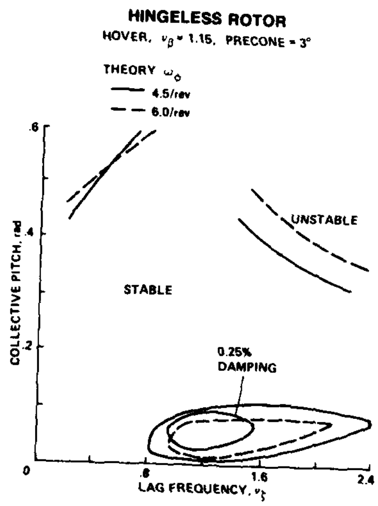

Fig. 28. Calculated flap-lag-torsion stability boundaries (Ref. 27).

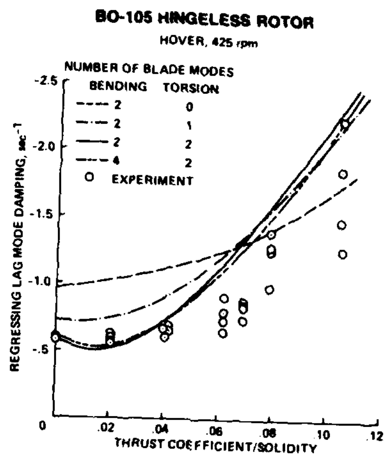

Fig. 29. Soft-inplane hingeless rotor stability (Ref. 30).

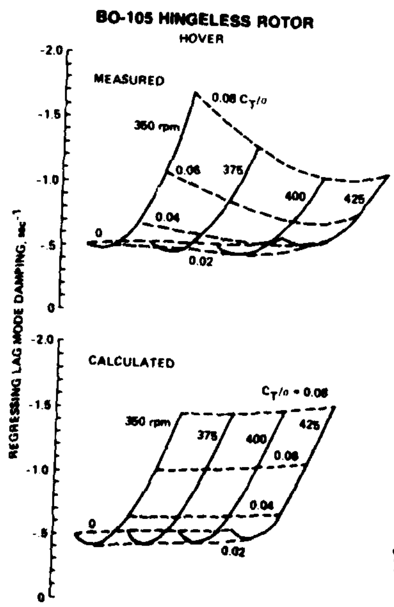

Fig. 30. Soft-inplane hingeless rotor stability (Ref. 30).

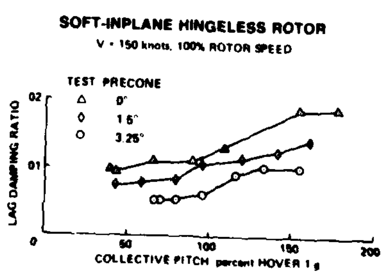

Fig. 31. Measured soft-inplane hingeless rotor air resonance stability (Ref. 13).

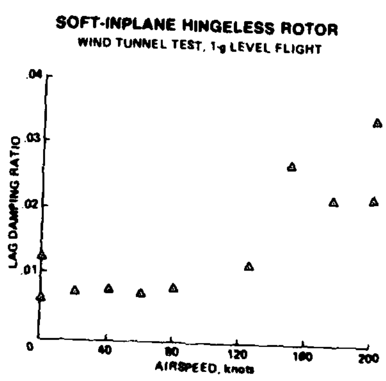

Fig. 32. Measured soft-inplane hingeless rotor air resonance stability (Ref. 13).

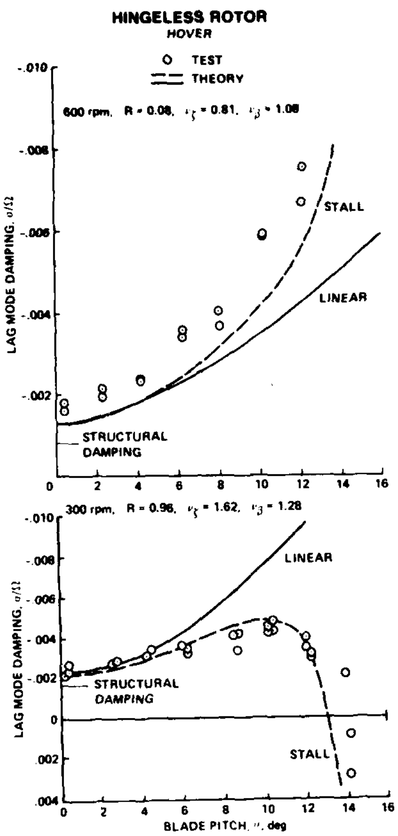

Fig. 17. Flap-lag stability produced by stall (Ref. 17).

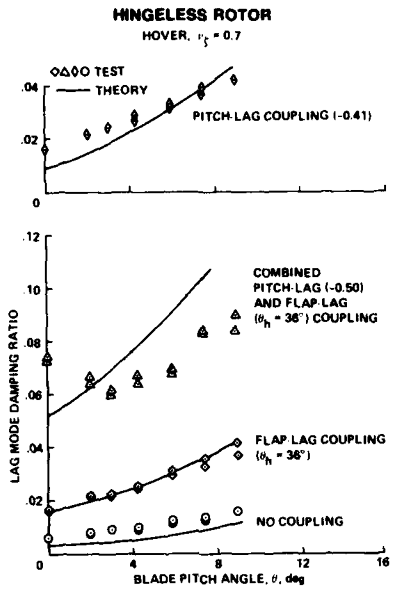

Fig. 18. Flap-lag stability (Ref. 18).

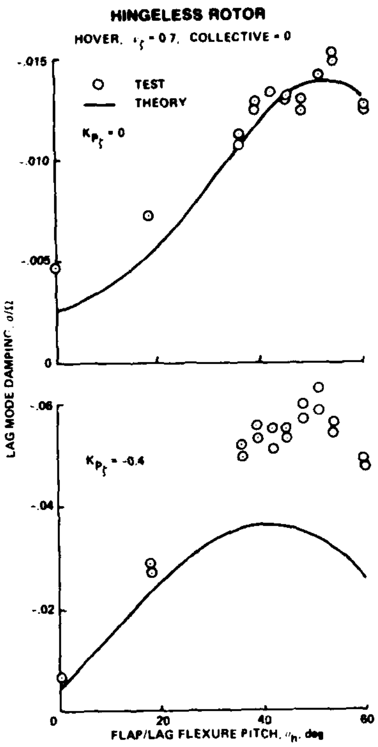

Fig. 19. Flap-lag stability (Ref. 18).

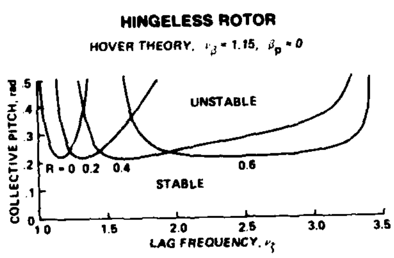

Fig. 20. Calculated elastic flap-lag stability boundaries (Ref. 24).

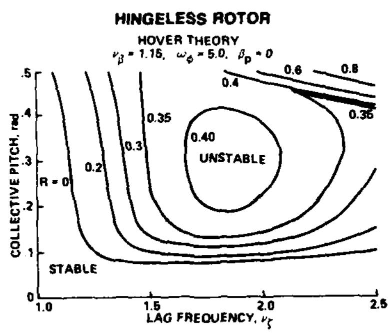

Fig. 21. Calculated flap-lag-torsion stability boundaries (Ref. 24).

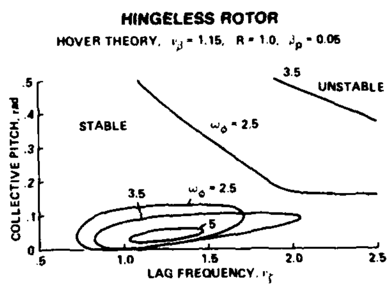

Fig. 22. Calculated flap-lag-torsion stability boundaries (Ref. 24).

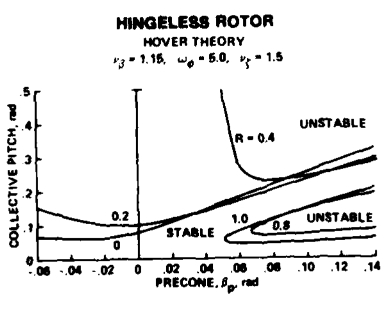

Fig. 23. Calculated flap-lag-torsion stability boundaries (Ref. 24).

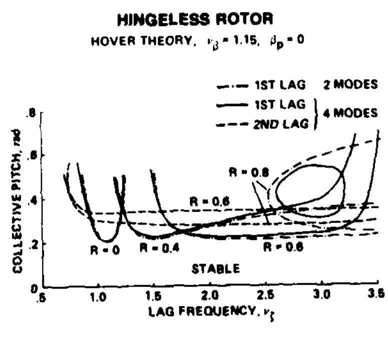

Fig. 24. Calculated elastic flap-lag stability boundaries (Ref. 25).

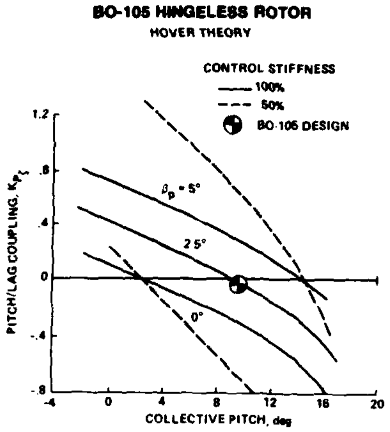

Fig. 10. Calculated pitch/lag coupling for BO-105 helicopter (Ref. 12).

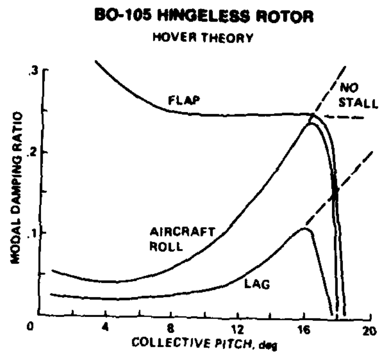

Fig. 11. Calculated flap-lag instability produced by stall (Ref. 12).

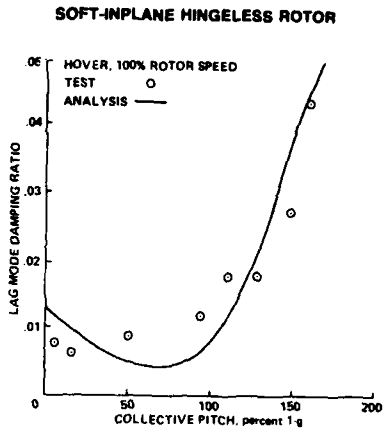

Fig. 12. Soft-inplane hingeless rotor air resonance damping (Ref. 13).

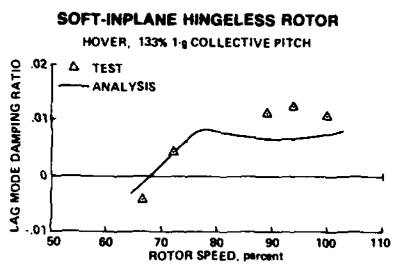

Fig. 13. Soft-inplane hingeless rotor air resonance damping (Ref. 13).

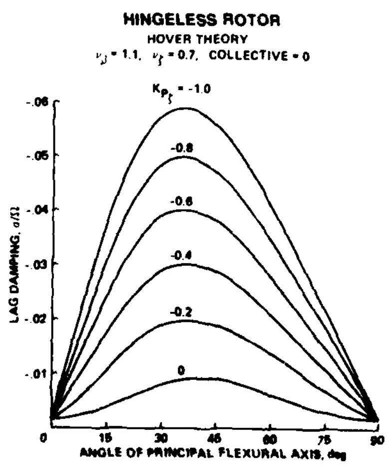

Fig. 14. Calculated hingeless rotor stability (Ref. 15).

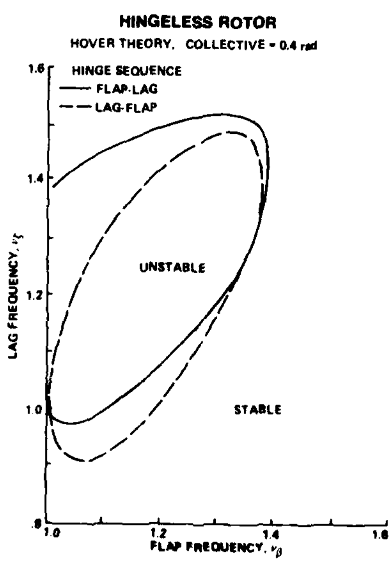

Fig. 15. Calculated flap-lag stability (Ref. 16).

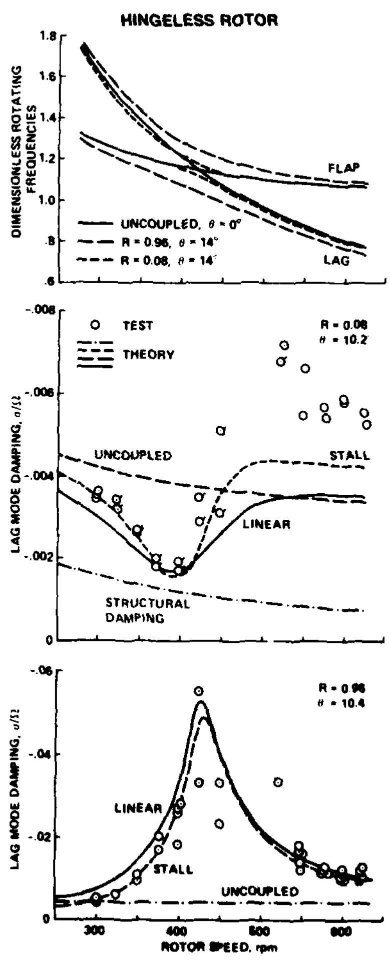

Fig. 16. Flap-lag stability (Ref. 17).

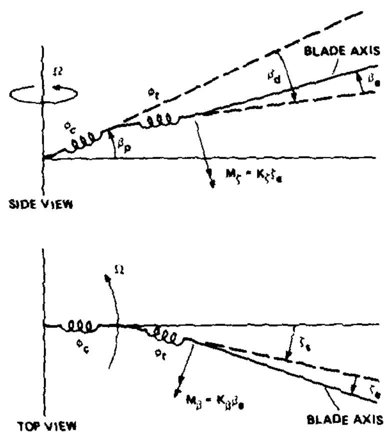

Fig. 1. Origin of effective pitch/lag and pitch/flap coupling.

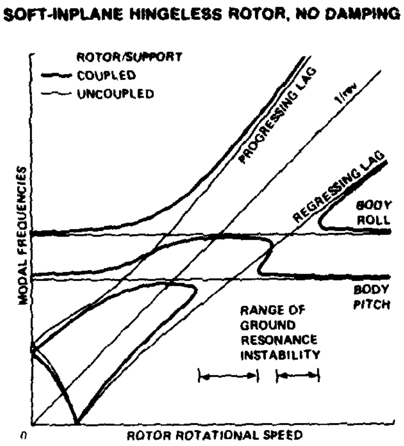

Fig. 2. Coleman diagram for ground resonance.

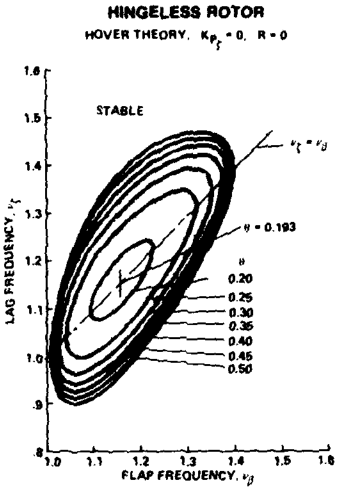

Fig. 3. Calculated flap-lag stability boundaries (Ref. 8).

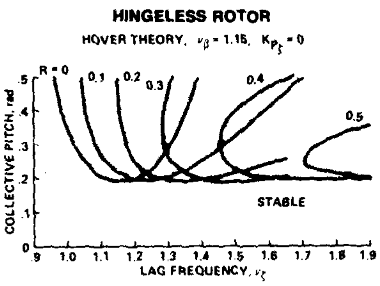

Fig. 4. Calculated flap-lag stability boundaries (Ref. 8).

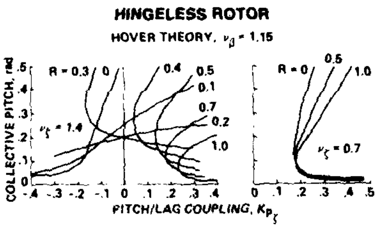

Fig. 5. Calculated flap-lag stability boundaries (Ref. 8).

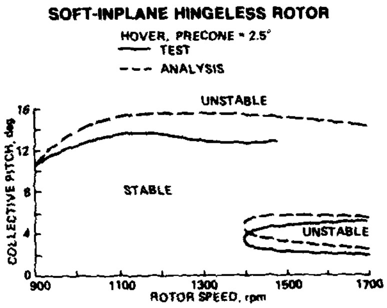

Fig. 6. Soft-inplane hingeless rotor stability boundaries (Ref. 10).

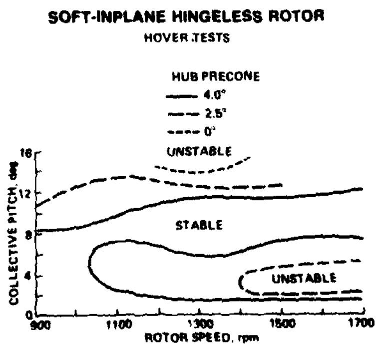

Fig. 7. Measured soft-inplane hingeless rotor stability boundaries (Ref. 10).

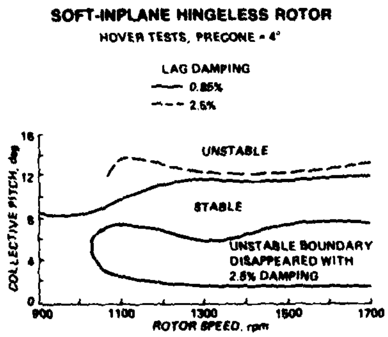

Fig. 8. Measured soft-inplane hingeless rotor stability boundaries (Ref. 10).

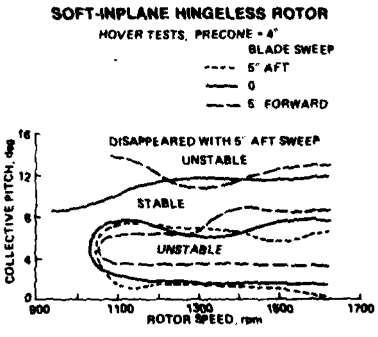

Fig. 9. Measured soft-inplane hingeless rotor stability boundaries (Ref. 10).

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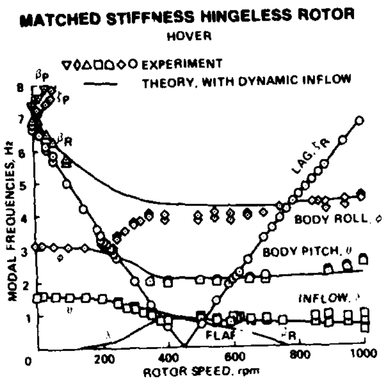

Fig. 85. Ground resonance stability with dynamic inflow (Ref. 80).

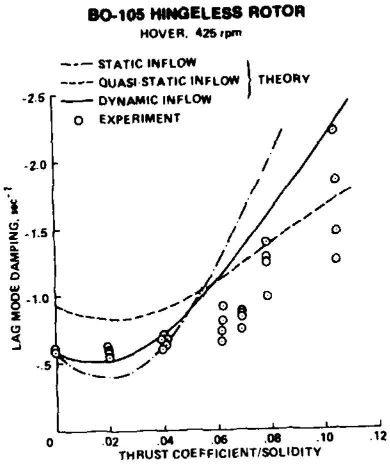

Fig. 86. Soft-inplane hingeless rotor stability with dynamic inflow (Ref. 30).

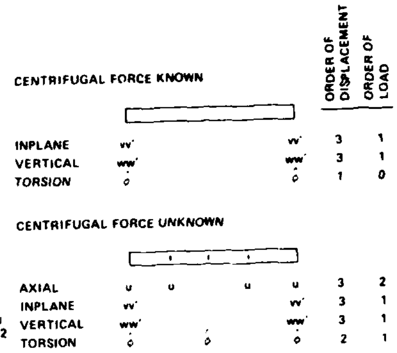

Fig. 87. Finite element beam degrees of freedom.

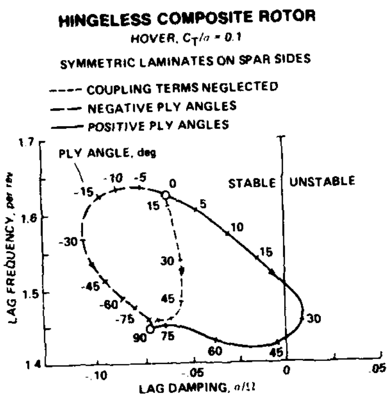

Fig. 88. Calculated hingeless rotor root locus with composite materials (Ref. 97).

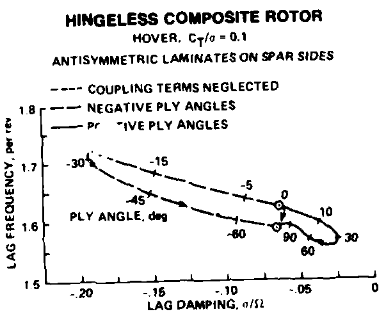

Fig. 89. Calculated hingeless rotor root locus with composite materials (Ref. 97).

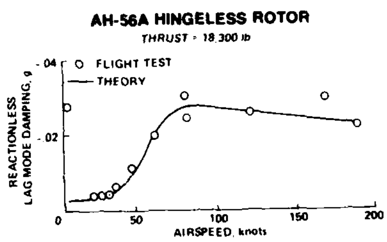

Fig. 90. AH-56A helicopter stability (Ref. 58).

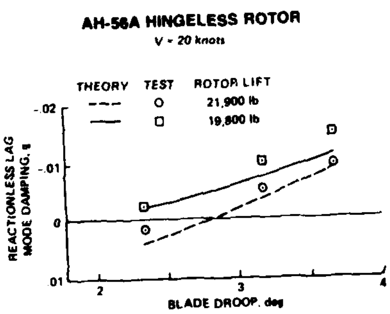

Fig. 91. AH-56A helicopter stability (Ref. 99).

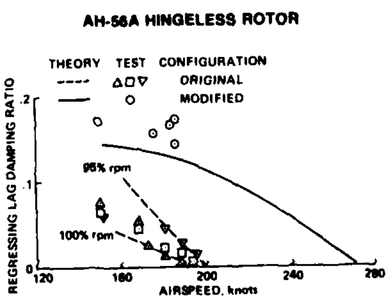

Fig. 92. AH-56A helicopter stability (Ref. 100).

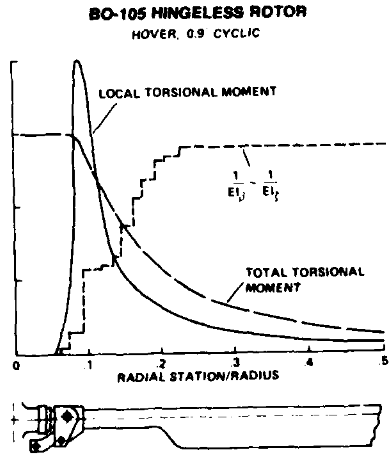

Fig. 93. Calculated BO-105 helicopter torsional moments produced by bending (Ref. 12).

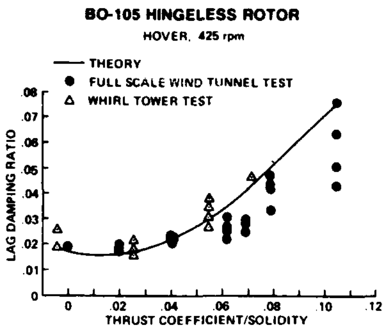

Fig. 94. BO-105 helicopter stability (Ref. 30).

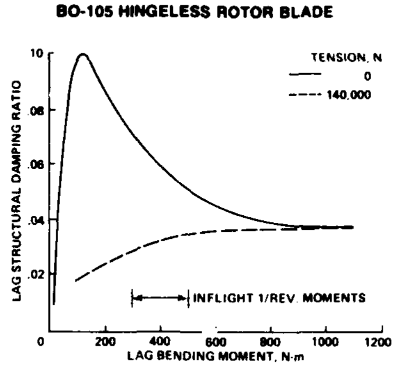

Fig. 95. Measured BO-105 blade structural lag damping (Ref. 98).

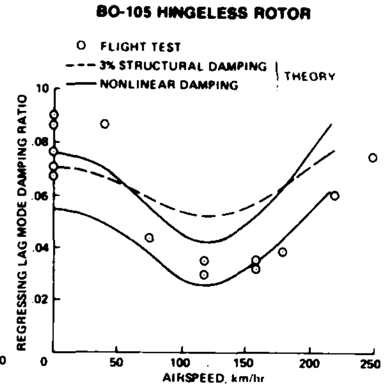

Fig. 96. BO-105 helicopter stability (Ref. 102).

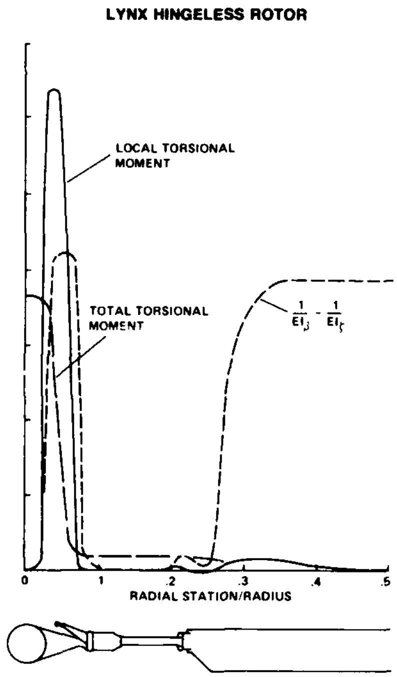

Fig. 97. Calculated Lynx helicopter torsional moments produced by bending (Ref. 14).

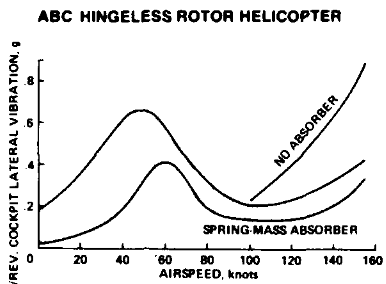

Fig. 98. Measured ABC helicopter vibration (Ref. 106).

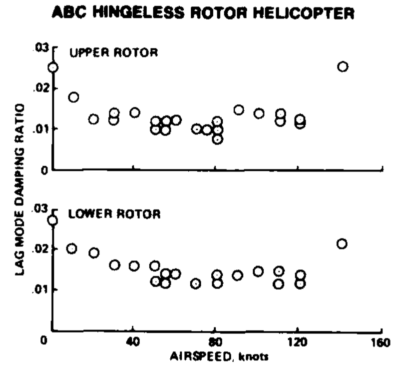

Fig. 99. Measured ABC helicopter stability (Ref. 107).

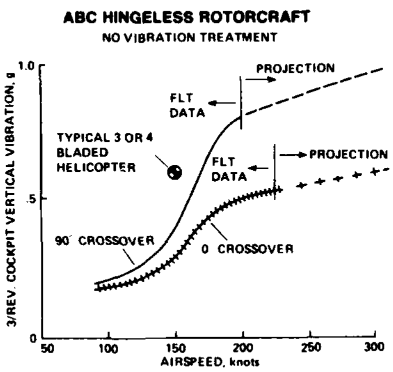

Fig. 100. Measured ABC rotorcraft vibration (Ref. 109).

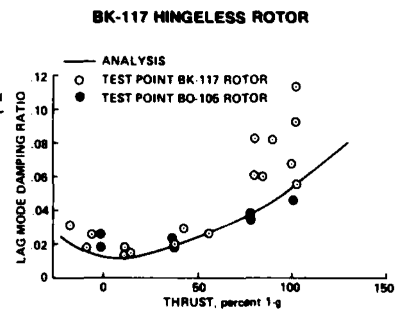

Fig. 101. BK-117 helicopter stability (Ref. 110).

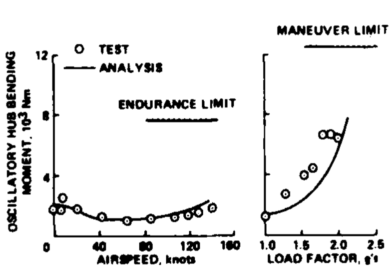

Fig. 102. BK-117 helicopter loads (Ref. 110).

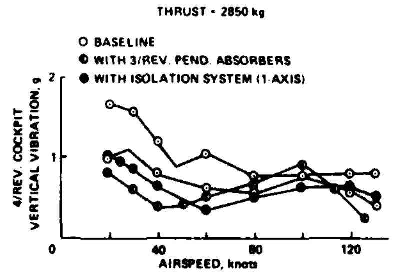

Fig. 103. Measured BK-117 helicopter vibration (Ref. 110).

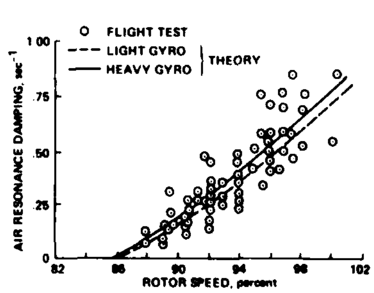

Fig. 104. XH-51A bearingless helicopter stability (Ref. 112).

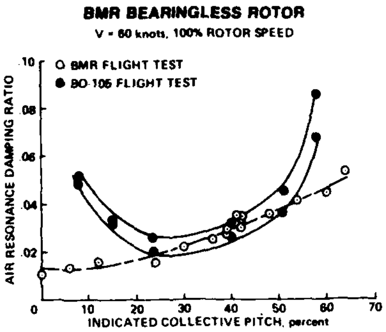

Fig. 105. Measured BMR bearingless helicopter stability (Ref. 113).

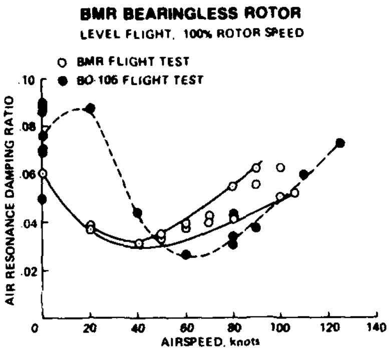

Fig. 106. Measured BMR bearingless helicopter stability (Ref. 113).

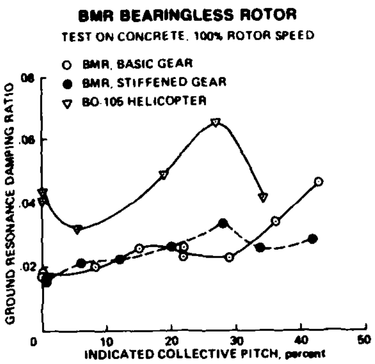

Fig. 107. Measured BMR bearingless helicopter stability (Ref. 113).

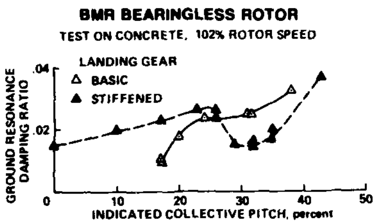

Fig. 108. Measured BMR bearingless helicopter stability (Ref. 113).

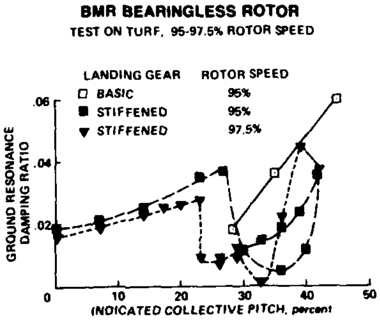

Fig. 109. Measured BMR bearingless helicopter stability (Ref. 113).

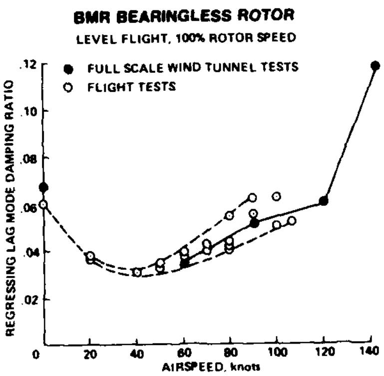

Fig. 110. Measured BMR bearingless helicopter stability (Ref. 41).

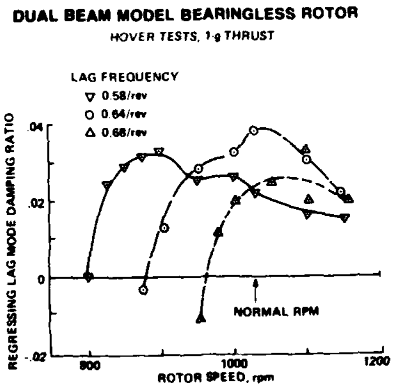

Fig. 111. Measured model bearingless rotor stability (Ref. 116).

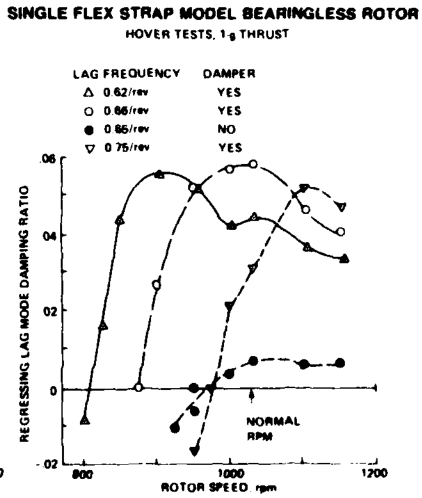

Fig. 112. Measured model bearingless rotor stability (Ref. 116).

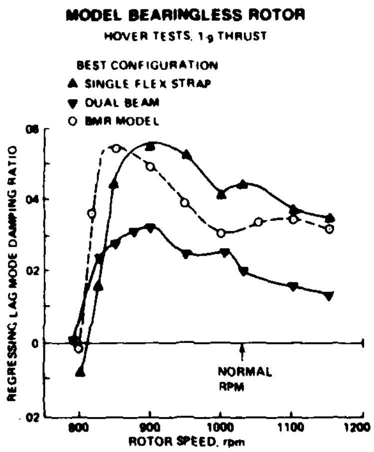

Fig. 113. Measured model bearingless rotor stability (Ref. 116).

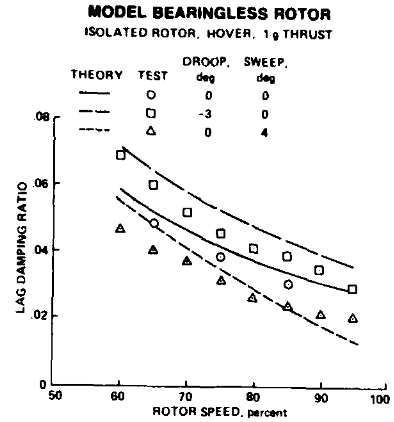

Fig. 114. Small scale Model 680 bearingless rotor stability (Ref. 119).

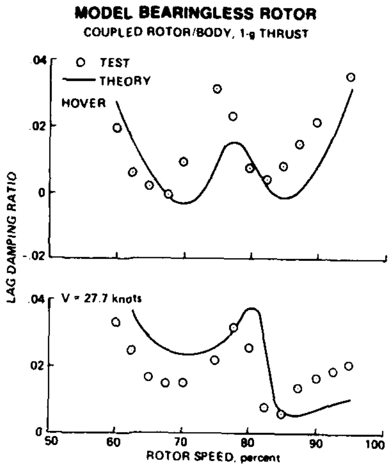

Fig. 115. Small scale Model 680 bearingless rotor stability (Ref. 119).

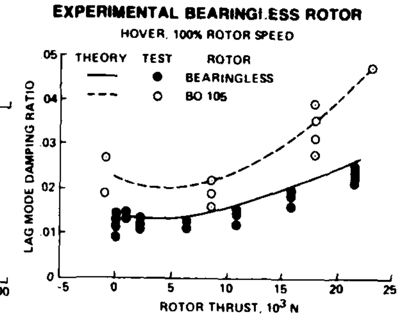

Fig. 116. Experimental bearingless rotor stability (Ref. 102).

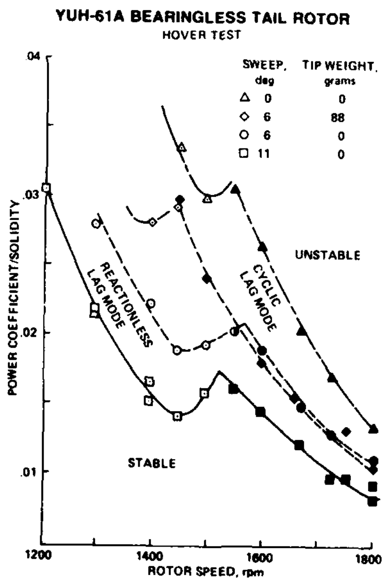

Fig. 117. Measured YUH-61A bearingless tail rotor stability boundaries (Ref. 123).

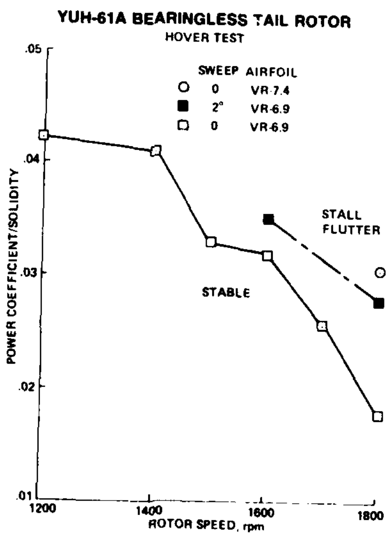

Fig. 118. Measured YUH-61A bearingless tail rotor stability boundaries (Ref. 123).

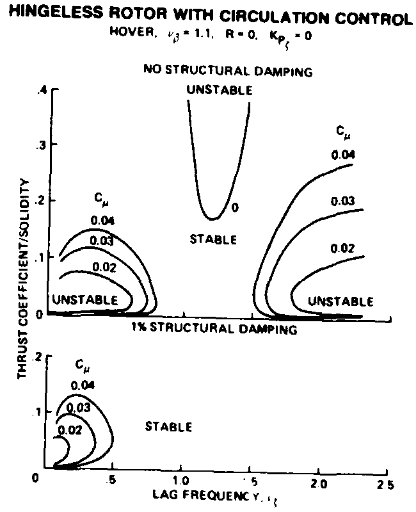

Fig. 119. Calculated flap-lag stability boundaries with circulation control (Ref. 127).

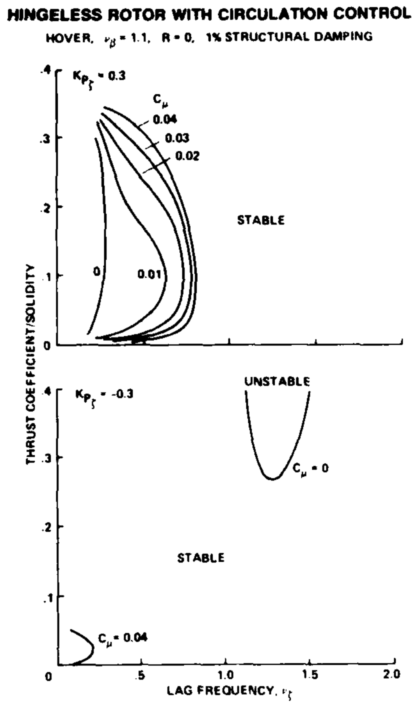

Fig. 120. Calculated flap-lag stability boundaries with circulation control (Ref. 127).

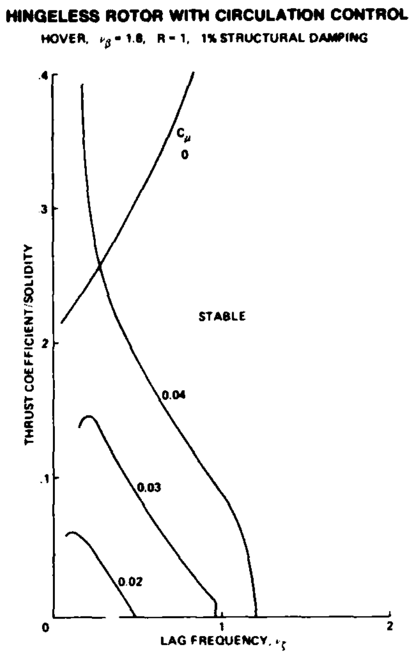

Fig. 121. Calculated flap-lag stability boundaries with circulation control (Ref. 127).

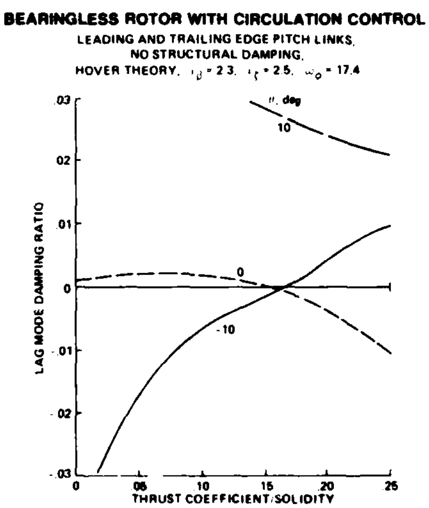

Fig. 122. Calculated bearingless rotor stability with circulation control (Ref. 45).

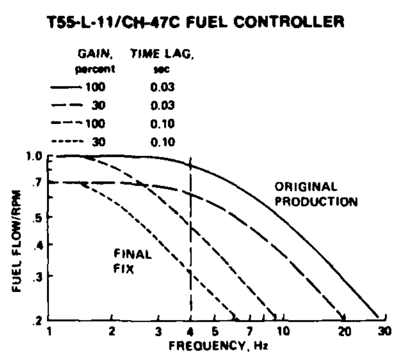

Fig. 123. Fuel control frequency response (Ref. 134).

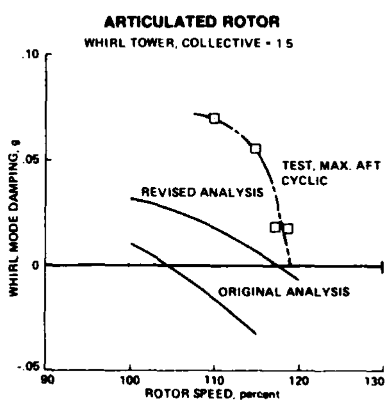

Fig. 124. Articulated rotor whirl mode stability (Ref. 137).

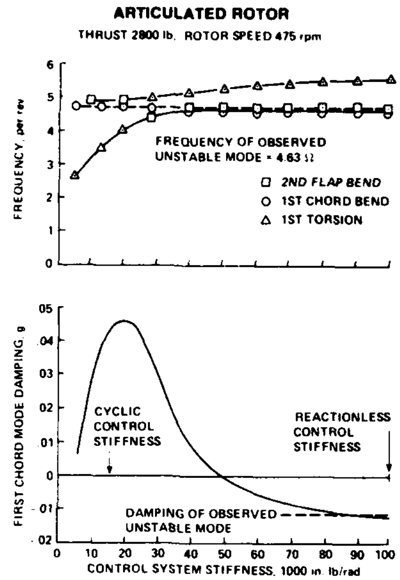

Fig. 125. Articulated rotor blade stability (Ref. 138).

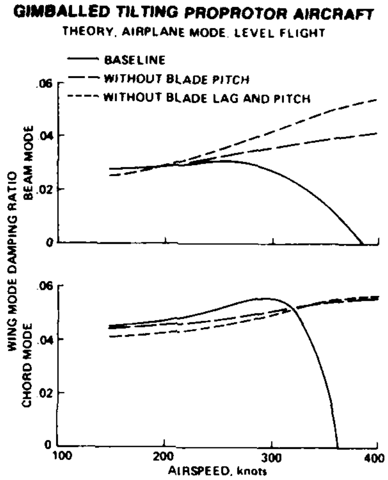

Fig. 126. Calculated tilting proprotor aircraft whirl flutter stability.

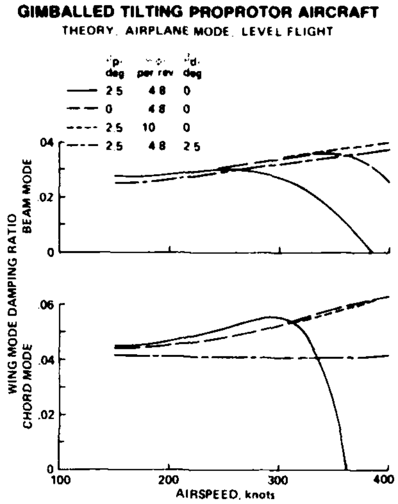

Fig. 127. Calculated tilting proprotor aircraft whirl flutter stability.

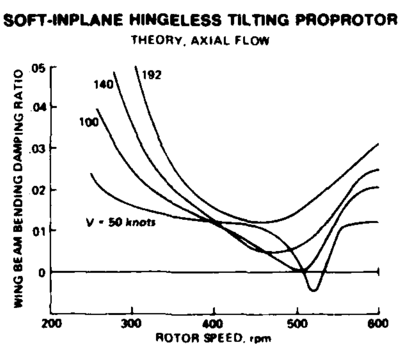

Fig. 128. Calculated tilting proprotor aircraft air resonance stability.

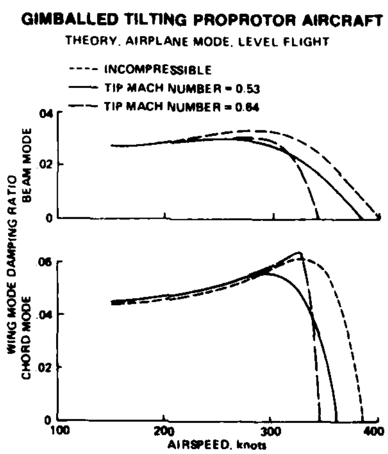

Fig. 129. Calculated tilting proprotor aircraft whirl flutter stability.

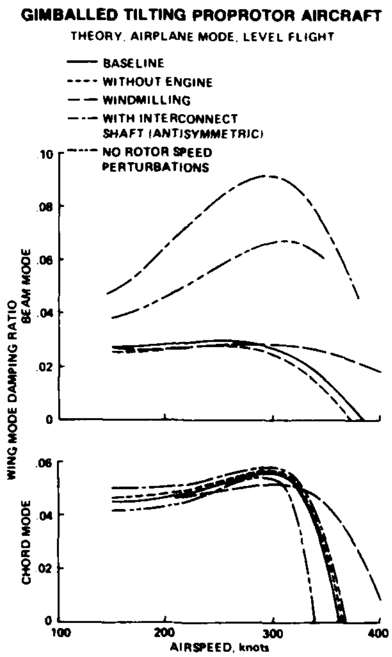

Fig. 130. Calculated tilting proprotor aircraft whirl flutter stability.

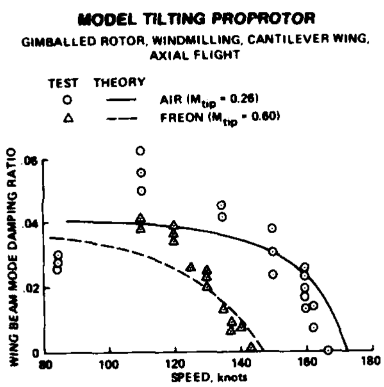

Fig. 131. Model tilting proprotor whirl flutter stability (Ref. 144).

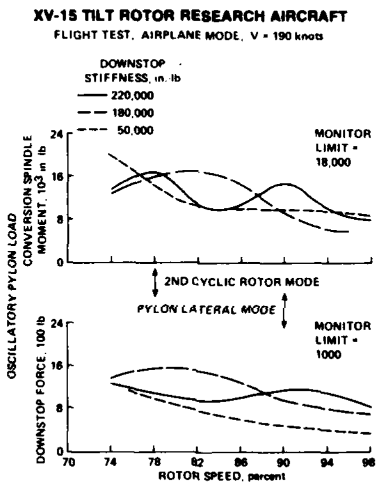

Fig. 132. Measured XV-15 Tilting Proprotor Research Aircraft pylon loads (Ref. 145).

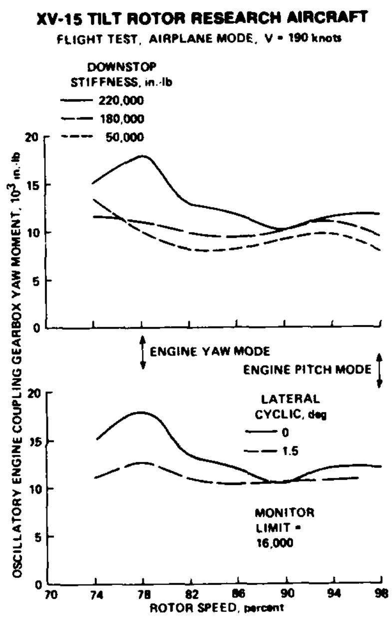

Fig. 133. Measured XV-15 Tilting Proprotor Research Aircraft pylon loads (Ref. 145).

MISSION REQUIREMENTS AND HANDLING QUALITIES

by

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SUMMARY

With the appearance of new missions for helicopters and with the development of a new generation of rotary-wing aircraft it has become obvious at the latest that future activities in the field of handling qualities must include the mission characteristics as well as the influences of the different subsystems implemented in the helicopter system. Therefore, mission analyses and consideration of system elements influencing mission performance are the basis for the lecture. The missions under consideration emphasize military missions but refer to civil missions, too. The system elements influencing mission performance include the basic helicopter, the pilot, the information system, the control system, interfaces, etc..

For evaluation of the overall pilot/helicopter system two different approaches are discussed, the evaluation of task performance and the evaluation of specific system characteristics resulting in handling qualities criteria. While task performance requirements may fulfill the orderers' demands, only specific system characteristics are useful as a design guide during helicopter development.

The status of handling qualities criteria is addressed including existing and proposed specifications. Shortcomings and critical gaps in the specification structures and especially in the data base are discussed. Ongoing activities in handling qualities research are directed (1) to the improvement and expansion of the data base considering present-day missions and technologies, and (2) to the establishment of new criteria considering the integration of different subsystems in order to improve the overall mission performance and to reduce pilot workload. These activities are closely connected with the availability of test facilities including ground based and in-flight simulators. The lecture concludes with a short overview on relevant research activities in the field of missionoriented handling qualities.

1. INTRODUCTION

For modern helicopters very high standards are required with regard to mission performance and system qualities. This is valid for both, civil and military aircraft. While for civil applications flight safety and profitability are the prime factors, the military users are asking in addition for adequate combat effectiveness.

The high demands on the helicopter are closely connected with the appearance of new missions expanding the role of the helicopter substantially. In civil IFR-operations the consideration of helicopter-specific capabilities and the reduction of weather minima for onshore and offshore operations are more and more required. For military applications specific missions have been defined, including low altitude operations in the entire speed range, night and adverse weather conditions, and hostile environment. The realization of these demands leads to an increasing complexity of the helicopter system and a constantly growing pilot workload, and copes the helicopter designer with the application of newest technologies. This includes, among others, the introduction of digital-electrical and -optical data transmission, of advanced sensors and display techniques, and of alternative non-metallic materials and respective design techniques.

The helicopter user is interested only secondarily in the arduous tasks which must be accomplished during the development process resulting from the integration of new technologies. He asks primarily for the demonstration of mission performance with consideration of acceptable pilot workload. Therefore, the helicopter user has to describe his planned missions in detail and to define his requirements with respect to flight safety and mission performance using quantitative and provable factors. While in the civil area this task is splitted between certification authority, being responsible for flight safety, and customer, the military procuring agency is responsible for the overall task.

The helicopter manufacturer needs complete and clear-cut requirements and criteria in order to be successful in the development and to be in the position to provide adequate data for the aircraft certification. In addition, the criteria may be used as a design guide helping to avoid a misleading development with reference to system performance and costs, even in an early stage.

Therefore, detailed requirements and criteria are the basis for communication between the customer, the certification authority, and the manufacturer. Airworthiness standards in the civil area and military specifications for military procurements are

normally used for this purpose.

The handling qualities are of decisive consequence for the intended use and the application of the helicopter, and hence they are an essential part of the requirements and criteria. These qualities or characteristics of the helicopter system describe its flightmechanical behaviour which enables the pilot to perform the tasks required. The requirements or criteria have to define corresponding limits in order to guarantee acceptable levels of pilot workload in all flight conditions under consideration.

The intention of this papar is to discuss (1) the contributions and influences of individual system elements of the pilot/helicopter system, (2) the evaluation of these elements in regard to mission performance and pilot workload, and (3) the establishment of handling qualities criteria.

2. MISSION ANALYSIS

2.1 MISSION AND MISSION PERFORMANCE

Unfortunately the term mission is used in several contexts and may actually have several meanings. Consequently a discussion about procedures of flying qualities evaluation has to start with fixing the terminology and the meanings of used terms. In Ref. 1 definitions of the terms role of an aircraft, mission, flight and flight phase are stated. They are largely quoted for this paper because they are clarified in close relation to flying qualities. The role of an aircraft defines its intended use in a general sense. The mission delineates this use in terms of specific objectives, that is, the required operations of the pilot-vehicle combination. The terms flight and flight phase denote the flight profile of a vehicle and its subdivision. Referring to this the set-up of a mission has to include the operational or mission requirements.

The primary aim of a user is to gain an optimal or at least high mission effectivity with a helicopter system in use or in acquisition. The effectivity is determined by the ratio of the attainable mission performance and the resulting costs. Mission performance summarizes the performance the pilot-helicopter system is able to accomplish in relation to the mission requirements. Figure 1 shows these relations with a brief pilot-in-the loop block diagram. The demanded mission can be transferred into commands for the pilot. He adapts his control strategy to obtain an acceptable vehicle response relative to the commands and tries to compensate deviations within his capabilities. Accordingly the resulting mission performance is mainly influenced by the characteristics of the helicopter system and the ability of the human pilot.

To get a satisfactory adaption of the technical systems to the mission objectives and to the human pilot the user has to determine and declare well-defined conceptions of the missions and the requested mission performance. Especially the interactivity of human pilot and technical system with the complexity of a man-machine system causes problems in establishing the mission requirements. Indeed, with support of modern technology it is possible to design helicopter systems which are easy to fly and which can perform new defined missions with the desired accuracy. But to avoid an explosion of costs a well-balanced compromise between desired mission performance and innovation of technology has to be found. Particularly high performance rotary wing systems must make full use of the pilots capability without overloading him or reducing necessary mission performance.

2.2 MISSION CHARACTERIZATION

A classical transport mission of a helicopter under visual meteorological conditions (VMC) can be characterized in adequate acceptance with the parameters speed, altitude and load factor. In operational flight envelopes the limits set by the mission are described. Figure 2 shows a possible envelope from Ref. 2. The required mission performance is to operate the vehicle in all states within the envelope boundaries without reaching rotorcraft limits and with a moderate pilot workload. This philosophy of defining requirements with respect to safety aspects is the base of civil flying qualities specifications (Refs. 3, 4). A more detailed definition of requirements can be indicated by the subdivision of the flight profiles in flight phases. In this way flight phases with specific requirements as instrument meteorological condition (IMC) phases or airwork phases are outlined (Fig. 3). A combination of flight phases in categories as terminal and non-terminal is obvious, if requirement statements can be referred to these categories. The current military helicopter specifications are based on this rationale. The MIL-H-8501 A uses a very rough subdivision into the hover and the forward flight region. Specific IMC requirements are added (Ref. 5). The MIL-F-83300 has a more systematic structure (Ref. 2). The requirements are organized with respect to classes of aircraft and categories of flight phases.

In the past decade the operational spectrum of helicopters has been expanded with vehemence for military use and for IFR operations, especially. The operational development includes an expansion of the helicopter role with new defined missions. Also it must be recognized that a specialization of demands exists for individual mission phases. Accordingly the flight envelopes and flight phases do not suffice to characterize mission demands. Figure 4 depicts briefly the requirements of selected missions. The general capabilities and the advantages of the rotory wing technique are fully utilized. Specific requirements on the manoeuvering performance are originated from flying close to the ground, using the terrain and ground obstacles to cover for protection, and contending in

combat situation. Phases of weapon delivery indicate demands on high preciseness related to the space position or flight path in combination with adequate tracking accuracy. Nap-of-the-earth (NOE) flight or air combat requires for excellent manoeuverability and high agility of the pilot-helicopter system.

To clarify the discussion of these mission attributes the following definitions for the terms manoeuverability and agility are offered:

- Manoeuverability is the ability of the pilot-vehicle system to change the velocity vector or the energy state. Manoeuverability can be measured and defined in the body fixed accelerations (n_x, n_y, n_z) and the rate of climb (Ref. 6).
- Agility is the capability to change the manoeuver state dependent on time. Agility can be measured and defined in earth fixed accelerations (Ref. 7). A good example of flight agility is felt to be performed by insects like the dragonflies. Staying in a hover position they can change very rapidly to another position followed by a high precision hover flight.

In order to establish mission oriented flying qualities it is necessary to subdivide missions in basic elements. An overview of rotorcraft mission elements is tabulated in Table 1. It should be noted that most of the elements in the flight phases low altitude, weapon delivery, and air combat are under discussion and are not defined as standards. Another objective must be to formulate flying qualities in dependance on the environmental situations. The tactical demand to perform military missions under reduced visibility, for example, can be accomplished with a reduction of the required mission performance, with a drastic increase of the pilots' workload, and/or with an intensified application of technology components to support the piloting task. With the many possibilities to relate the requirements to mission elements and graduated environments the necessity exists to find a structure which reduces and organizes the high number of requirements. The US Army and Navy have initiated a systematic effort to revise the current rotorcraft specification. Two proposals have been developed which cover the above mentioned topics of requirements organization.

The rational behind the proposed revirement of Ref. 8 is to categorize the basic elements of operational missions. In the next step of the specification methodology the assumption is made that a clear coordination of mission elements and necessary rotorcraft response characteristics can be defined (Tab. 2). A hierarchy of the required response types is indicated from an acceleration response type to a translational command system. Also the visual cue environment situation is integrated in the scheme of response types. This approach represents an important principle of the specification structure, because the reduction of requirements related to only one parameter is effected. The required response type must be upgraded in the presence of degraded usable cue environment (UCE) according to Table 3. The usable cue environment is defined in a scale between 1 and 5. For more details see section 3.3.

The approach of Ref. 9 has a structure using a characterization of mission requirements that is divided in two classification schemes. The mission are divided in segments (here named flight phases) and can be characterized by the characteristics: manoeuvering required, precise flight path or space positon required, and target tracking required (Tab. 4). With these characteristics eight flight phase categories can be defined as combinations of the possible characterizations that are related to differences in the mission-related requirements. To specify requirements in accordance to the visual cue a matrix scheme in proposed which describe four elements of required operational capability (Tab. 5). This approach of operational characterization yields a reduction to maximum elements of 32 to which the specification has to be arranged.

3. SYSTEM ELEMENTS INFLUENCING MISSION PERFORMANCE

The final objective to procure an aircraft is the achievement of satisfactory performance in the aircraft's mission. The pilot will be able to achieve this performance if the aircraft has suitable handling qualities. The usual definition for handling qualities is (Ref. 1): "Those qualities or characteristics of an aircraft that govern the ease and precision with which a pilot is able to perform the tasks required in support of an aircraft role." This definition includes many subject areas of the pilot/aircraft system such as task, instrument display and avionics sophistication, aircraft stability and control characteristics, aircraft configuration, pilot stress, workload, tracking performance, environment, failure modes, and others. To illustrate the relationship of the primary elements of the pilot control loop to the operation of the pilot/vehicle combination Figure 5 shows the influencing factors. With so many variables and the special characteristics of helicopters, the subject of helicopter handling qualities becomes a very complicated one. The assessment of the handling qualities of the complete system requires not only the knowledge of the characteristics of the isolated subsystems including the pilot, moreover the system integration is the decisive factor. Therefore handling qualities can be determined best by actual experiments with the pilot in-the-loop and through the use of pilot opinion and rating systems.

The following section will present a short review of the fundamental characteristics of the most essential subject areas influencing pilot/helicopter system performance.

3.1 BASIC DYNAMICS OF HELICOPTER FLIGHT

In this part of the lecture some of the basic control and stability characteristics of the helicopter will shortly be reviewed (Refs. 10, 11, 12).

Fundamentals of Rotor Control

The classical helicopter rotor has its blades hinged close to the center of rotation so that they are free to flap (and lag) in accordance with the aerodynamics and mass forces acting on them (Fig. 6). Hingeless rotors, although having no flapping hinge, are conventionally described as being equivalent to articulated rotors but with the addition of a flap hinge spring restraint. Similarly its blades flap in order to maintain equilibrium between the aerodynamics, mass and elastic forces.

Rotor control consists of varying the blade pitch angles in a collective and cyclic manner: the feathering motion. This causes a rotor thrust vector which can be controlled in its magnitude and in its direction by the appropriate choice of the blade pitch angles. In hovering flight collective pitch - applied equally to all blades - determines only the magnitude of rotor thrust, whereas cyclic pitch - varying in a sinusoidal fashion with a frequency of once per rotor revolution - leads to a once-per-revolution flapping motion of the blades which may be considered as a tilt of the rotor disc relative to the shaft. As the rotor thrust vector remains essentially perpendicular to the disc it is clear that a disc tilt results in a tilt of the thrust vector. In forward flight the control of thrust vector magnitude and direction is coupled because of the changes in blade lift due to different flow at the advancing and retreating blades.

The resulting moment at the rotor hub is in the same direction as the disc tilt and is proportional to this tilt. The hub moment per unit disc tilt is on the other hand proportional to the offset of the flapping hinge from the center of rotation, to the blade flapping inertia, and to the hinge spring stiffness corresponding to a hingeless rotor.

Thus the total moment about the helicopter center of gravity produced by the rotor consists of two components: the stiffness contribution resulting from hinge offset and spring-induced hub moment, and a moment caused by the component of the thrust normal to the shaft and in the direction of tilt. Figure 7 (Ref. 13) shows how the total rotor control moment is splitted up for two typical rotor systems. For the articulated rotor the thrust vector tilt is the main component and the stiffness component may equal, at most, the thrust vector component. It is obvious therefore that the control moment about the helicopter center of gravity of articulated rotors is highly dependent on the value of the rotor thrust and consequently may vary considerably within the flight envelope. However the hingeless rotors of today produce control moments primarily by stiffness and the total control moment is about 3-5 times higher than for articulated rotors. Although the thrust vector tilt component is much the same as for arcticulated rotors its contribution is only about 25% of the total which implies only a small variation of control power with rotor thrust level (Fig. 8). Any external disturbance to the rotor can be looked at in terms of effective control inputs so that in a very crude manner the rotor moment per unit cyclic pitch can be considered as an indication of the sensitivity of the rotor to external disturbances.

As discussed above, the feathering motion of the blades including cyclic and collective blade pitch angle variations provides a very convenient means for inclination and magnitude adjustment of the rotor thrust vector. The system commonly used to control the desired blade pitch angles at the rotating blades is a swashplate assembly consisting of a rotating and a non-rotating part as shown in Figure 9. In pitch motion the blades are constrained by a linkage connecting them to the rotating part of the swashplate assembly. From the non-rotating part of the swashplate control rods lead off through the control transmission to the pilot's stick and collective lever. By this system the swashplate can be tilted and moved vertically and hence produces a pitch change at the rotating blade in reference to the rotor shaft.

Helicopter Control Characteristics

The control of the helicopter rigid body motion in general requires the possibility to influence three forces and three moments corresponding to the six degrees of freedom (three translatory and three rotatory DOF) in space. These components in the body fixed axis system with the origin at the helicopter center of gravity are: longitudinal, lateral, and vertical forces and roll, pitch, and yaw moments (Fig. 10). By means of the rotor control, i.e. inclination and magnitude adjustment of the rotor thrust vector, the pilot is able to control five components: longitudinal force, lateral force, vertical force, roll moment, and pitch moment. Due to the characteristics of rotor control longitudinal force and pitch moment as well as lateral force and roll moment are connected or coupled with each other. These components can not be controlled independently so that the pilot can manage the control of five components using only three controllers:

- stick fore and aft for longitudinal force and pitch moment,
- stick sideward left and right for lateral force and roll moment, and
- collective lever up and down for vertical force.

The sixth component, the yaw moment, is usually provided by the tail rotor. By using

the pedals, the pilot can alter the collective pitch of the tail rotor blades and - due to the resulting tail rotor thrust change - he can control the yaw moment.

In order to limit the loads in the rotor blade root area, in the hub, and in the shaft and the vibratory inputs from the rotor into the fuselage of the helicopter, it is necessary to minimize the rotor disc tilt required for helicopter trim. This in consequence limits the center of gravity offset from the rotor shaft line of single rotor helicopters to about 2.5% of the rotor radius. This requirement for low levels of disc tilt within the flight envelope entails the use of considerable cyclic pitch control as illustrated in Figure 11. The diagram indicates the stick displacements required for a specific helicopter in the speed range for zero tilt of the rotor disc. Obviously the helicopter fuselage and tail-plane aerodynamic characteristics and any center of gravity offset necessitate some disc tilt for trim purposes. In addition, there is also a requirement for control displacements for manoeuvers and some allowance must be made for coping with emergencies such as a failure in any autostabilisation equipment. The final pitch requirements for a helicopter tend therefore to be higher than indicated in Figure 11.

Although at the higher speeds the locus of trim cyclic requirement is predominant in one direction this is definitely not so at the low speed around hover. Furthermore the locus of trim cyclic stick is not indicative of the control displacements required to change from one flight speed to another. Only when trimmed flight at the desired forward speed is attained the cyclic pitch position will coincide with the trim cyclic curves. In short, control requirements change in a most complex manner during manoeuvers.

As mentioned above, the helicopter controls are coupled and the magnitude of these couplings vary with flight conditions. In Figure 12 this behaviour is shown for a typical hingeless helicopter (Ref. 14). At high speeds the initial pitch acceleration due to collective control input has nearly the same magnitude as due to cyclic control input, which is the primary control. In order to effect a compromise between various aspects of coupling it is common practice to introduce some compensation between the controls in the mechanical linkage system. But in general, it is impossible to design mechanical interlinkages in such a way that the control behaviour is improved at all points of the flight envelope. So, in spite of the good controllability of modern helicopters, the pilot's workload is high due to the necessity to respond to undesirable behaviour produced by the control characteristics of the helicopter. This is true especially in very demanding missions requiring rapid manoeuvering in different speeds and with precise flight path control.

Stability Characteristics

The stability analysis of an aircraft deals with its dynamic behaviour after being disturbed in its initial trim conditon by any disturbance like a gust or a control input. The initial tendency of the aircraft to return to its trim condition is called static stability. The dynamic stability deals with the oscillation of the aircraft following the disturbance from trim. Because the mathematical theory of helicopter stability is rather complex and lengthy, the following discussion will only consider some of the most significant aspects.

Static stability is related to factors being very essential for the pilot, like stick displacements required to change speed and to change the normal load factor. As can be shown by a detailed analysis the moment change at the center of gravity due to an attitude or angle-of-attack change is of great influence for the consideration of helicopter static stability. This so called angle-of-attack or attitude stability includes contributions from the rotor, the fuselage, and the horizontal stabilizer, as shown in Figure 13 (Ref. 13). The fuselage contributes normally in an unstable sense whereas the tailplane of course, as in fixed-wing aircraft, has a stabilizing effect. For many reasons, helicopter tailplanes usually cannot be made large enough to provide overall stability. The rotor contribution to the angle-of-attack derivative is normally in an unstable sense and depends to a high degree on the flapping hinge offset or the stiffness of the rotor. In addition, this effect depends on the collective pitch of the rotor causing a different behaviour of the helicopter operating at a high or a low collective pitch like in autorotation. The variation of the moment at the center of gravity due to changes in translational speed is a measure for another contribution to static stability. This so called speed stability derivative normally tends to reduce the speed and provides therefore a stabilizing share to static stability. Because several, in some cases counteracting contributions are involved in helicopter static stability the determination of overall stability is very complex and requires a detailed analysis of even small effects in the total speed range. In Figure 14 a procedure for determining static (stick position) stability using flight tests is presented (Ref. 15). As can be shown, positive static stability is a necessary but not a sufficient requirement for dynamic stability.

For helicopter dynamic response studies, discussion is normally limited to control-fixed time history responses after a disturbance from a trimmed flight condition. Free control-motion information usually centers on control system effects like control friction, breakout forces etc., because most helicopters are equipped with irreversible, power-boosted control systems. The control-fixed dynamic modes of longitudinal motion of a helicopter in forward flight are typically characterized by a stable oscillatory or non-oscillatory short term motion and a possibly unstable low frequency oscillatory motion, the phugoid motion. The characteristics of the phugoid motion are mainly determined by the above discussed angle-of-attack and speed derivatives and in addition by the pitch damping being attributable to pitching angular velocity and producing a stabilizing mo-

ment opposite to the pitching motion. Pitch damping can be strongly increased by rotor characteristics like the Lock number (ratio of aerodynamic and inertial forces) of the rotor blades and the increase of rotor stiffness, as shown in Figure 15 (Ref. 16).

In Figure 16 (Ref. 17) the influence of rotor stiffness on helicopter longitudinal dynamic stability is demonstrated. Starting from unfavourable stability characteristics in hovering flight the low stiffness rotor helicopter (Ω_β = 1.01) is more and more governed by the stabilizing horizontal tail. With the increase in rotor stiffness a destabilizing trend due to the contributions of the angle-of-attack stability can be observed. The introduction of stiffer rotors on high speed helicopters is seen to have exacerbated the stability problem, at least in the controls-fixed (open loop) case. Of course stiffer rotors have considerably more control power available and the overall stability characteristics with pilot reaction included (closed loop) may well be superior to those of articulated rotor helicopters.

Apart the stability characteristics of longitudinal and lateral motion (not discussed here) the strong coupling of the responses (and control inputs) further complicates the prediction of the dynamic stability. As shown in Figure 17 a longitudinal stick input to a typical hingeless helicopter in level flight produces a pitch rate, of course, but in addition a roll rate of quite the same magnitude and a rather high yaw rate response. The inertial cross couplings (roll-due-to-pitch and pitch-due-to-roll) depend upon various rotor system parameters with the most important being Lock number and rotor stiffness, as shown in Figure 18 (Ref. 16).

The comments made above only refer to general but most important helicopter characteristics and set the scene and provide a brief insight into the complicated character of helicopter dynamics.

3.2 STABILITY AND CONTROL AUGMENTATION SYSTEMS

The goal of the helicopter designer is to make his aircraft have such good inherent characteristics that the pilot requires no extra help. The fact that so many helicopters are flying today with just a pilot in control demonstrates that the goal can be reached, if expectations are moderate. For many applications, however, such as flying in IFR missions or while the pilot has other demanding tasks it is essential to reduce pilot workload and to improve the performance of the overall system pilot/helicopter.

Many forms of stability augmentation systems and automatic stabilization equipment have been utilized to improve the basic handling qualities of helicopters by enhancing, modifying, and improving the angular damping, control and manoeuvering response, stability, and long-term trim as well as providing relief for the pilot. Two primary feedback quantities are utilized for the basic functions of these systems. They are attitude and attidude rate. For more sophisticated coupled operations, radio position signals (ILS, VOR etc.), radio-ratè signals, barometric signals, radar signals (radar altimeter), airspeed data, accelerations, heading etc. may be added to the basic function of the automatic stabilization equipment.

In the past different strictly mechanical systems were used, providing a limited degree of artificial stability. In these devices, used in Bell, Hiller, and Lockheed helicopters, a mechanical gyro is an integral part of the rotor system producing a mixture of rate damping and attitude stability for a certain frequency range (Ref. 15). In modern helicopters such mechanical devices are no longer used because they introduce additional mechanical complexity and the improvement in stability is combined to a certain degree by a deterioration of the control characteristics. The effects and use of these devices are widely documented in literature (Refs. 11, 12, 13).

Electromechanical systems offer, on the other hand, a much more flexible means for stability and control improvements and can obviously be accommodated completely within the airframe. The following discussion will be limited to electromechanical systems considering only the most basic types and functions of these systems. They are (Ref. 15):

- Stability Augmentation Systems (SAS),
- Stability and Control Augmentation Systems (SCAS), and
- Automatic Stabilization Equipment (ASE).

A SAS is any system that enhances or supplements the angular damping and the stability of the basic aircraft. Usually both attitude and rate signals are used as feedback quantities requiring an attitude (vertical) gyro type system. With relation to the helicopter, a new or "synthetic" aircraft stability derivative is created and provides an attitude stability with respect to the horizontal flight path. Since the helicopter can provide stability only with respect to speed so that use of this "new" stability derivative is an extremly important addition to the system and is essentially the foundation for single-pilot IFR flight in helicopters. Because many basic helicopters represent low damped systems in vital portions of their flight envelope, the use of rate with attitude feedback is also very important. Use of just attitude signals or rate signals alone usually cannot optimize the stability of the aircraft for all conditions of the flight envelope.

A SCAS is any system that enhances, modifies or supplements the stability, angular damping, and control system response. Use of the term SCAS normally implies that feed-forward loops are utilized and pilot's control motion is sensed. Figure 19 shows the basic difference of SAS and SCAS systems together with a simplified block diagram of a typical SCAS. In SCAS systems the augmentation actuators will move either with the pilot's control input or against it in order to establish or maintain a pitch (or roll) rate that is proportional to his input. For example, if a gust upsets the aircraft and the pilot does not move his controls, the augmentation actuators tend to suppress or attenuate the effects of the gust.

Both SAS and SCAS systems are considered pilot-in-the-loop or hands-on systems. When attitude signals are used in these systems they are usually "washed-out" or "leaked-off" in the short term and this tend to provide an attitude type response only in the very short term and a rate type response in the longer term.

The most basic function provided by Automatic Stabilization Equipment or Autopilots is the attitude-hold mode. Those functions will maintain aircraft pitch, roll, and yaw (heading) long-term automatically, for pilot-out-of-the-loop, hands-off type operation. During manoeuvering flight with the attitude-hold system engaged the pilot is actively interrupting the attitude-hold feedback loop on a moment by moment basis and placing a new reference attitude on the autopilot. When the aircraft is being actively manoeuvered by the pilot with the attitude-hold loop of the autopilot disabled, he may be operating a vehicle that has either the basic characteristics of the helicopter or the characteristics of an augmented helicopter with artificial augmentation of some kind installed (e.g. SAS or SCAS) and operating full time. Although this attitude-hold systems provide a high degree of flight control workload relief for the pilot, they may not provide it long term, because they do not include attitude and airspeed-hold which may cause severe problems depending on the specific characteristics of the helicopter. In order to achieve real autopilot-type functions, it is usually necessary to provide additional data as feedback information and close the loops. This may include information from installed radio navigation and approach systems (VOR, ILS, etc.) providing automatic guidance, tracking, and flight path steering. Such a system is frequently referred to as a fully coupled autopilot.

The application of feedback systems for helicopters is more and more increasing. This is caused on the one hand by increased mission demands, especially for IFR operations, on the other hand by the availability of the controller technology. For military applications and for large helicopters the extent of automation is usually higher than for civil applications and small helicopters. Table 6 presents some features of the control system of modern helicopters. Figure 20 (Ref. 18) shows a typical automatic stabilization equipment of a modern military helicopter (AH-64).

The application of non-redundant electrohydraulic system in the helicopter control system asks for intensive consideration of possible failure conditions. The worst case normally is a failure referred to as actuator "hardovers". With a high-gain, full-authority actuator system a hardover can present very dangerous situations to the pilot if the failure occurs at an inopportune time (as it is usually the case).

Because of the hardover problem and its serious consequences to the control of the helicopter, system augmentation must be accomplished either through the use of limited-authority actuators for high-gain systems or low-gain, low run-rates for full-authority actuator systems. To date almost all systems make use of high-gain, limited-authority series-type actuators. Limited-authority is used in the sense that most modern high-gain actuators move the swashplate only about 5 % to 15 % of its full motion. The term series-actuators is used in the sense that it denotes an inner-loop augmentation system where the pilot's control stick does not move as it does in parallel or outer-loop systems. Overside capability of all augmentation and autopilot systems is always afforded to the pilot.

As a result of the problems discussed above, some compromises must be made that ordinarily would not have to be made with a full-authority and redundant system. In order to achieve all possible improvements, the artificial means installed in the helicopter may comprise fly-by-wire or fly-by-light logic schemes, high-gain and low-gain, full-authority and limited-authority, series and parallel systems and actuators, programs and logic to decouple the helicopter controls and responses. This means might require automatic failure diagnosis, monitoring, and correction systems that would allow to operate the helicopter with a high level of reliability and safety. In Figure 21 the Advanced Digital Optical Control System (ADOCS) is shown in its basic elements (Ref. 19). The primary system function is to replace the mechanical control linkages of todays helicopters by optical signalling. The computerized system is able to provide among others the integration of different automatic flight control modes, of multiaxis sidearm controllers, and of an integrated helmet and display sight system. The application potential of such a system includes essential contributions for the improvement of the efficiency of the total helicopter system. Among these are:

- increase of mission effectiveness,
- reduction of pilot workload, and
- improvement of reliability.

It is assumed that those systems will be available for operational use in near fu-

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many institutions having potential to contribute to this overall objective: universities, military and civil authorities, flight test centers, and national research institutions. Only a concentration of all activities will yield adequate coverage of handling qualities procedures and criteria with general acceptance and in acceptable time. Of course, the industry also is treating problems in this area using excellent facilities, but their objectives are substantially directed towards the determination of configurational rotary-wing parameters to meet defined requirements than to define the requirements. The motive may be that the industry has to be more interested in the development and design of specific helicopter systems rather than in the generation of data for general purpose. Nevertheless any support of industry is worthwhile to overcome the lack of data.

In consequence of the test facilities being available, the research institutions especially are put in for the position to contribute above all to the generation of handling qualities data having the required generality. The specific test facilities with relevance to the mentioned objectives are listed in Table 9. More details are provided in Ref. 70.

7. CONCLUDING REMARKS

In this lecture an overview of handling qualities aspects of modern helicopters was given. It was discussed that the evaluation of the pilot/helicopter system has to include all factors influencing the closed loop system behaviour. These factors are: the mission, the pilot, the cockpit interface, the aircraft characteristics, the aircraft environment, and the mission performance.

For evaluation of the complete pilot/helicopter system two different approaches have been outlined, the task performance evaluation and the evaluation of specific system characteristics. The discussion of the existing handling qualities criteria showed substantial short-comings whereas the development of new criteria is restrained in particular by critical gaps in the handling qualities data base.

In this situation the research activities in this area have to be increased and coordinated, including all organizations and institutions having corresponding potential and test facilities like ground-based and in-flight simulators. Only these cooperative efforts will yield adequate handling qualities specifications for the helicopter development programs under discussion in different nations.

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In this situation, there is a need for a more systematic approach to the specification structure and for an expanded data base on which to base new requirements. Therefore, the US Army and Navy in 1982 have initiated a systematic effort to develop a new generic specification for the handling qualities of military rotorcraft. It was stated that other NATO countries are interested to cooperate in the establishment of this specification in order to fill the gap in this area. The motivation for these activities is increased in addition by ongoing or planned military helicopter projects in different nations. These programs include the JVX joint services vertical lift aircraft and the Army LHX advanced scout/attack/utility helicopter programs in the US, and the PAH-2/HAP/HAC multirole attack helicopter and NH 90 battlefield transport helicopter programs in Europe.

Experience in previous efforts to revise the handling qualities specification showed that the primary obstacle to developing new requirements was a lack of systematic data from which new general criteria could be developed and substantiated. Therefore, it seems to be necessary to coordinate and to encourage the efforts of all organizations having potential research and test activities in this field. Only this concentrated effort will possibly overcome the enormous data gaps.

Phase 1 of the program to establish a new specification was directed towards the development of a new structure and was finished in 1984. It is assumed that the first version of the specification ready for adoption can be finalized not earlier than end of 1986. At this time not all of the above mentioned data gaps will be closed, so it will be necessary in future to gather additional data from several sources like development programs and experimental research activities using operational helicopters and ground-based and in-flight simulators.

6. ACTIVITIES IN HANDLING QUALITIES RESEARCH

In the previous sections the data gaps have been addressed and necessary activities have been mentioned in order to resolve the gaps. They can be summarized and adjusted to the following areas of endeavor:

- definition of mission elements,
- derivation of flight test tasks representative for the mission elements,
- determination of task performance procedures, and
- quantification of task performance parameters.

Especially for the evaluation of rotary-wing system characteristics the following necessary efforts have to be performed in addition:

- determination of procedures for level 1 requirements,
- quantification of level 1 requirements, and
- degradation of requirements for level 2 and 3.

In response to these needs extensive research activities are required including theoretical investigations and experimental programs in order to establish a viable data base which adequately covers all the stated topics.

One main objective of the theoretical investigations is the extension and improvement of mathematical rotorcraft models being a prerequesite for conducting simulation tests with good validity. For these activities relevant data of analytical studies, wind tunnel test programs, and system identification approaches from flight tests are utilized.

For experimental research programs different facilities can be identified with potential for rotorcraft flying qualities research:

- ground simulators,
- in-flight simulators, and
- operational helicopters.

These facilities have substantial consequence for the development of evaluation procedures and evaluation quantification. Because the pilots' ratings and comments are used as basic data for a correlation analysis with objective test results a high demand of simulation validity and environment reality is required. Therefore, operational helicopters are used to cover the determination of task performance procedures and parameters, whereas the simulation test facilities are in urgent need if variations of rotary-wing characteristics are required. In addition the operational helicopters are worthwhile tools to verify the fidelity of simulation results in specific test points, the so-called anchorpoints.

Generating a viable data base asks for a coordination of the efforts of all organizations with activities in the area of helicopter flight mechanics research. There are

pling response criteria can not be covered significantly with only one parameter, because the coupling response over the frequency range of interrest at least is influenced by the coupling in the controls and the cross-coupling, both in relation to the primary axis responses. Nevertheless, Figure 47 shows a recommendation diagram with test results of Refs. 63 and 64. The influence of the pitch to roll axis coupling is outlined with the cross-coupling derivative related to the roll damping. It is felt the diagram gives a clear impression of the high influence of coupling on the pilot ratings.

All procedures described in this section, to define handling qualities criteria demonstrate possible approaches and are not generally accepted. In comparison to the fixed-wing criteria the helicopter handling qualities are just in the beginning to adapted to new implemented technologies and to new established operational demands. A high number of critical gaps exist which have to be filled for the establishment of substantial rotary-wing handling qualities requirements. The helicopter handling qualities research is asked to generate a viable data base which can be used for the definition of mission-oriented handling qualities criteria.

5. STATUS OF HANDLING QUALITIES SPECIFICATIONS

In this section of the lecture a short review of some existing and proposed handling qualities specifications will be presented, following recent publications in this area (Refs. 59, 66, 67, 68, 69). The discussion will be concentrated on military specifications because of the current and worldwide activities in this field.

With the definition of new helicopter missions and with the introduction of new technologies in the development, it has become apparent that the present helicopter handling qualities specification, MIL-H-8501 A, cannot accurately assess the characteristics of these aircraft. The fact that MIL-H-8501 A was last updated more than 20 years ago only tends to amplify this point. Even in flight phases covered by this specification like level flight in VMC, essential criteria like dynamic stability criteria, are no longer valid for modern helicopters in general.

There are three main deficiencies of MIL-H-8501 A:

- The criteria are not mission-oriented. The specification uses for example a weight parameter for hover control power considerations that is the result of scaling laws and is not meant to represent the variations in control response which may be required for mission differences. The criteria are basically for VMC flights with only unsatisfactory consideration of IMC requirements. The need to perform increasingly complex missions in adverse weather and at night asks for the definition of criteria considering all of the pilot's tasks involved in the mission together with an integrated treatment of vehicle dynamics, flight control system characteristics, cockpit controllers, displays and vision aids.

- The specification delineates no systematic breakdown into hover/low speed and forward flight criteria. Many single rotor helicopters show a coupled pitch-roll dynamic oscillation in hover, whereas in forward flight a dutch-roll type response is often found. In order to address this different axis couplings, a breakdown of the specification into hover/low speed and forward flight criteria would be meaningful. In addition the control characteristics in the two speed regimes are quite different.In hover a helicopter pilot tends to use longitudinal, lateral, and directional controls independently, whereas in forward flight the pilot needs to use lateral and directional controls in a coupled manner similar to fixed-wing aircraft pilots.

- MIL-H-8501 A does not include quantitative levels of degraded handling qualities for specific failures. The specification has qualitative criteria for failures of power boosted controls, automatic stabilization systems, and engines.

There have been several formal attempts to revise MIL-H-8501 A. One major effort was the development of the V/STOL specification MIL-F-83300, which incorporated all the data available at that time and followed closely the structure and format of the specification for fixed-wing aircraft (MIL-F-8785 B). This specification attempted to include both, helicopters and fixed-wing V/STOL aircraft. Although MIL-F-83300 has never been used in the procurement of a new airframe, several potential weaknesses have been identified, resulting mainly from the fact, that this specification is primarily based on V/STOL data and covers only lightly explicit helicopter characteristics. For example, the requirements were divided into hover/low speed (less than 35 knots) and forward flight (35 knots to V_{CON}), with the idea to convert at V_{CON} to the fixed-wing requirements of MIL-F-8785B, being too rigorous for helicopters. In addition, no systematic consideration has been made of helicopter-specific missions. Nevertheless, a comparison with MIL-H-8501 A shows that MIL-F-83300 has clear advantages in its broad coverage of important handling qualities aspects and its systematic structure.

For recent procurements in the US (UH-60A and AH-64) a specific set of handling qualities criteria was developed and incorporated into the so-called Prime Item Development Specifications (PIDS). The real test for a specification, how well it does in providing the desired helicopter, showed significant handling qualities improvements over previous helicopters. However, it may be very difficult and sensitive to assess the specification's merit in this connexion.

by including a delay in the equivalent system model. Figure 43 shows the results of applying this procedure to the hover roll attitude command transfer function of the VAK-191B VTOL aircraft (Ref. 59). The procedure includes to define different criteria for different response types. A proposed classification scheme is detailed in Ref. 56 and is based on an extensive compilation of VSTOL vehicle dynamics. For example the proposed LOS form for both attitude and attitude rate augmented vehicle response is given by the equation

$$\frac{\text{attitude change}}{\text{cockpit control deflection}} = \frac{K(1 + 1/T)e^{-\tau s}}{(s + \lambda)(s^2 + 2\zeta\omega s + \omega^2)}.$$

For attitude response, the LOS form reduces to

$$\frac{\text{attitude change}}{\text{cockpit control deflection}} = \frac{Ke^{-\tau s}}{s^2 + 2\,\zeta\omega s + \omega^2}.$$

The delineation between the two general types of response is provided by the time domain criterion of Figure 44. If the impulse response of the system lies within the drawn boundaries it must satisfy the criteria for an attitude system. If it violates the boundaries at any point, it is considered to be a rate system. If the response type is classified and its LOS equivalent is determined, related criteria are used to assess the acceptability of the dynamics. Figure 44 also shows proposed criteria for rate and attitude response system. The same procedure has to be applied for additonal response types like translational rate command and the other response axis. Equivalent systems are a well accepted concept for defining level 1 flying qualities for helicopters. However, the complexity of the response of unaugmented helicopters due to inter-axis coupling, makes it unlikely that useful equivalent system forms of sufficient generality can be defined for the level 2 and 3 boundaries.

An approach which does not assume a particular form of response is the bandwidth criterion. Most familiar handling qualities metrics are, in fact, related to bandwidth. The bandwidth describes in the open-loop frequency response the ability of the pilot to close the loop. The bandwidth procedures are transformed from pilot-in-the-loop modelling techniques (Ref. 60). A recent utilization of bandwidth was in the Neal-Smith criterion, which proposed an evaluation for good closed-loop pitch tracking for fighter airplanes (Ref. 61). From the pilots point of view, a high-bandwidth response would be described as "crips" or "rapid and well damped". Typical commentary for a low bandwidth response might be "sluggish to control input" or "tends to wallow". The evaluated response also includes the coupling behaviour the pilots have to compensate. This overall evaluation is the main advantage of the bandwidth response. Hence it may be suited for helicopters, where coupling tends to mask the classical response forms. In Ref. 55 results of variable stability in-flight simulations are discussed which indicate that the bandwidth hypothesis is indeed valid i.e., the coupling itself matters only to the extent that it affects bandwidth. A definition of bandwidth for handling qualitiy purposes is given in Figure 45. The bandwidth is the frequency at which the phase margin is 45 deg or the gain margin is 6db, whichever is lower.

As the helicopter became more complex many handling qualities researchers felt that the time history contained the parameters influencing the pilots' evaluation. Especially the helicopter designers prefer a formulation of criteria in the time domain, because most of the helicopter modelling for design evaluation is done in the time domain. The formulation of handling qualities criteria in the time domain also is recommended in the proposed structures for the MIL-H-8501 A revision (Refs. 8, 9). As an example, charactezing parameters for a rate and attitude response are shown in Figure 46. The parameters are a dead time, an effective risetime, and an amplitude ratio of the first cycle with respect to the steady state value. These parameters can be directly related to parameters describing the frequency response. Once formulated as shown, however, the requirements are independent of system's order and apply directly to the actual step response - thus avoiding problems of interpretation. Boundaries for level 2 and 3 also can be defined using the same parameters and including the coupling behaviour.

Inter-axis coupling is well recognized as one of the most severe handling qualities problem with unaugmented rotary-wing aircraft. Fixed-wing aircraft tend to be much less affected by such coupling with the exception of direct force controlled aircraft and of high angles of attack manoeuvers. Accordingly the shaping of coupling is under discussion for the fixed-wing handling qualities. In a coordinated turn study an evaluation approach is recommended to evaluate the combined use of aileron and rudder. Ref. 62 indicated that the response to rudder inputs necessary to coordinate turns plays a dominant role in evaluations, and proposed a quantitative measure of acceptable and unacceptable characteristics. The recommended criteria are based on parameters which characterizes the rudder crossfeed and the magnitude of rudder required. With the combination of both parameters a characterization of the crossfeed frequency response can be obtained.

In helicopter handling qualities studies the shaping of response behaviour has often been assessed with the derivatives of the linearized model of motion (Refs. 53, 63, 64, 65). This can be viable parameters to define required controllability. Especially cou-

Another approach is proposed in Ref. 54. Tasks like slaloms or sidesteps with emphasis on the inner-loop control of bank angle can be subdivided in a series of discrete bank commands. Modes during limited intervals are examined with the technique of phase plane trajectories as shown in Figure 38. The effective task performance can be expressed by the commanded net bank angle changes and the corresponding peak roll rates during that changes. The crossplotted parameters illustrate the agressiveness of the pilot-helicopter system. Two modes of pilot operating conditions seem to be involved in the range of roll flight tasks. For small-amplitude modes the magnitude of peak roll rate is about proportional to the bank angle command values. For large-amplitude modes the peak roll rate is fairly independent of the bank angle commands. Recommended boundaries can be drawn of which the lower boundary characterizes the requirement of the task and the upper boundary represents the capability of the helicopter system. A similar technique is discussed in Ref. 48 for an evaluation of mission requirements.

4.2 EVALUATION OF SYSTEM CHARACTERISTICS

As mentioned, the more worthwhile approach for the helicopter design phase is offered by specifying flying qualities in parameters which characterize helicopter dynamic behaviour, the handling qualities criteria. That means, test results performed from the overall system have to be reduced and related to the control elements and the controlled element of the pilot-in-the-loop feedback system. It must be recognized that a baseline helicopter is much more complex than for example a fixed-wing aircraft. The inherent asymmetry of single-rotor helicopters causes them to have several features that complicate analysis and specification of handling qualities criteria. There is a strong coupling between the longitudinal, vertical, and lateral-directional response. They are highly nonlinear. They inherently involve more than the classical rigid-body modes used to represent the responses of conventional fixed-wing aircraft. The complexity increases with implemented augmentation and artificial information systems. Therefore the necessity is to determine a viable mapping of the high order system into a lower-dimensional form with a minimum of variables suitable for flying qualities specification (Fig. 39). A possible structure is to associate the criteria with the controller and their primary response. Additionally coupling and control harmony criteria have to be involved.

Historically, the handling qualities of rotary-wing aircraft have been vastly inferior to their fixed-wing counterparts. For example, the pitch attitude control of most operational helicopters will not even meet the level 3 requirements of military fixed-wing aircraft. Level 3 is defined as a Cooper-Harper rating of worse than 6-1/2. An example is illustrated in Figure 40 from Ref. 55. Interestingly, the pilot ratings from many previous helicopter studies indicate that rotary-wing pilots are willing to accept much less than the fixed-wing community. This is shown in Figure 41 with data from Ref. 56. The helicopter pilots rated configurations with a damping ratio lower zero with satisfactory whereas the fixed-wing specification requires for level 2 (corresponds to ratings of 3-1/2 to 6-1/2) a damping ratio better than zero. This is felt to occur for two reasons: (1) helicopter pilots are trained to cope with, and expect as normal severe instabilities and cross-axis coupling, and (2) the tasks used in the evaluations were not sufficiently demanding. In previous years the tasks used in experiments to obtain handling qualities pilot ratings allowed the pilot to compensate the system deficiencies due to the low task demanding. The recent operational missions become increasingly severe to the limits of pilots capability and for that marginal handling qualities can no longer be tolerated. It is therefore not unreasonable to expect the same quality of response in helicopters that is currently enjoyed by fixed-wing pilots. In fact, the agile and precise manoeuvering in some NOE mission elements may make it necessary to impose more stringent requirements than are necessary for fixed-wing aircraft. But nevertheless it is a helpful approach to verify the adaptability of techniques to define handling qualities criteria used by the fixed-wing community for helicopter application (Ref. 57).

Many modern helicopters employ a stability augmentation system. The implementation of feedback technology will be considered with increase of operational demands. In consequence, the failure situations have to be taken into considerations in handling qualities criteria. The concept of levels is used in MIL-F-8785C to specify the allowable degradation in handling qualities in the presence of failures. The level 1 corresponds to a satisfactory pilot rating and is required for the normal situation. For failures states level 2 or 3 is required which correspond to ratings of acceptable and unacceptable. The specification of level 2 and 3 handling qualities will tend to be more critical in rotary-wing aircraft. This is a result of the relative poor handling qualities of the baseline helicopter without augmentation and hence the possible large change in the dynamics before and after a failure. This is illustrated in Figure 42, which shows a dramatic shift in the characteristic modes after a SAS failure in the CH-53D helicopter.

The criteria can be formulated in the frequency or time domain. The advantages of the approaches shall not be argued in this paper. The importance to be considered in both efforts is to come up with only a few numbers of parameters which fit the high order dynamics of the rotary-wing system. In frequency domain the MIL-F-8785C defines lower order equivalent systems. The matching technique is outlined in detail in Ref. 58. Briefly, the approach used is to match the frequency response in amplitude and phase of the high order system (HOS) over a given frequency range with a preselected low-order system (LOS) model which minimizes the cost function defined by the equation where gain

$$M = 20/n \sum \left((\text{gain (HOS)} - \text{gain (LOS)})^2 + 57{,}3\,(\text{phase (HOS)} - \text{phase (LOS)})^2\right),$$

is in db and phase is in radians. Large amounts of high frequency lag are accounted for

For the both flying qualities methodologies an extended data base must be set up. As the pilot is the essential scale for a quantitative classification of the evaluation parameters it is necessary to provide realistic conditions to him. So flying qualities research especially requests for the use of suitable test facilities:

- operational helicopter,
- in-flight simulator, and
- ground-based simulator.

The different areas of application exploit the advantages of the individual facility (Fig. 31). The operational helicopter is prefered for tests, if excellent fidelity of reality is requested: (1) assessment of mission requirements, (2) derivation of representative flight test tasks, and (3) definition of task performance parameters. In-flight and ground-based simulators are qualified for studies to determine handling qualities criteria which require the ability to vary the characteristics of system dynamics.

4.1 TASK PERFORMANCE EVALUATION

The first step in all efforts to define flying qualities parameters is the derivation of test tasks for the flight or simulator test approaches. The flight test tasks shall represent the requirements of specific mission elements and shall be defined in considerable detail, including specific control techniques and performance limits. For the helicopter projects UTTAS and AAH the US-Army has specified a vertical displacement terrain avoidance manoeuver (Refs. 45, 46). Figure 32 illustrates the flight task and shows for comparison time histories of the AH-64 helicopter from Ref. 47. The time histories outline the fulfillment of required performance limits. But nevertheless an indication of the workload of the pilot and a relative evaluation cannot be drawn only by verifying the required performance limits. Involving pilots in a test approach and requesting for pilot ratings and comments suggest an essential aspect for the definition of test tasks. Establishing flight test tasks the complexity of the mission elements has to be reduced to get clear relations between pilots' evaluations and test results. This aspect is important for the test pilot and the test engineer, because the pilot should be enabled to evaluate and comment the test results and the test engineer has to interprete pilot ratings and test results in a correlation analysis. An often realized characterization of the forward flight close to the ground is a dolphin and slalom task, which divide the more-axis manoeuvering in a vertical and lateral task. Figure 33 shows slalom and dolphin task from Ref. 48.

Ref. 31 has specified the height and time over the dolphin obstacles as the most important task performance evaluation parameters. Depending on the control activity, measured in the longitudinal and collective controls, boundaries for the exposure area can be drawn which correlate very well with the pilot ratings (Fig. 34). Additional parameters can be identified, the maximum pitch attitudes and maximum load factors, which can increase the pilot workload. Results of tests with varied control strategy yielded the recommendation, the dolphin can be performed more effectively with emphasis on the collective control than using primarily the longitudinal control. But flying vertical displacement tasks with primary use of collective results in more severe requirements for the vertical control dynamics and control response (Ref. 49).

One of the most essential question of todays military missions is the necessary performance in lateral manoeuvering. Different simulation and flight test studies have been conducted to investigate the influences on lateral task performance. At RAE a triple-bend task for simulator test approach was composed (Ref. 50). The accuracy of the triple-bend track was stated to be the main parameter describing the performance for this task. In addition the amount of control activity points to an increase of task difficulty. Figure 35 depicts the differences in control activity depending on the speed, whereas the track achieved at different speeds is practically indistinguishable. The necessity in general to combine task performance and pilots' control activity is also underlined in Ref. 51 which describes the results of hovering tracking tests.

At NASA and DFVLR similar slalom courses have been constructed. Both courses are representing very well the requirements for the roll axis in NOE operations. Besides the seperately performed flight tests a test program was jointly planned and conducted to compare the test techniques and to explore influences on task performance (Ref. 52). The changes in roll rate seems to be the main influence describing the agility demands of a slalom task and in consequence correlates very well with the evaluations of the pilots. Figure 36 gives an impression of this characterization using the half power bandwidth of the roll rate (Ref. 53). If the overall pilot-helicopter system is able to react on the ground track commanded bank angles with quick roll rate changes the task performance is improved and the pilots tend to a better rating. In general, a slalom task shall be performed with primary use of lateral control. In steady turns the relation between commanded bank angle and load factor can be expressed as $n_z = 1/\cos \phi$. This relation can also be taken as a reference for a slalom task and the amplitude of deviations describe a deterioration of task performance (Fig. 37) (Ref. 48). Two main influences substantially stand for the deviations measured in the tests: (1) If the roll capability of the helicopter is insufficient the pilots in addition have to use side slipping to maintain the track. (2) High coupling behaviour in roll to pitch and roll to heave also is an influence factor, which increases the pilot workload and debases the task performance, if the pilot is not able to compensate.

ples is an extremely tough problem to deal with, especially since it is virtually impossible to design a large asymmetrical airframe with sizable access doors and large mass concentrations without having several modes of response close to any given rotor excitation frequency.

In Figure 29 the vibration levels are shown as measured in flight tests for the helicopter BO 105 (Ref. 42). Besides the vertical cabin vibrations in the total level flight speed region the frequency spectrum for 30 and 100 kt speeds are presented. It is obvious that the essential contributions come from frequencies determined by the product of blade number (4) and rotational frequency of the rotor (Ω = 7 Hz) and its multiples. These curves are typical of the vibration characteristics of a helicopter in its different flight speed regimes. It can be seen that there are two regimes, low speed flight and high speed flight, where the vibration levels are especially high.

In the past, flights at cruising speed were the primary conditions considered decisive in verifying vibration levels since flight at this speed was usually the longest flight mode. However, the advent of terrain flying as an accepted doctrine in military missions makes reevaluation of vibration requirements in low speed manoeuvering flight an absolute necessity.

Comparison of current and past helicopter human factor requirements applicable for all steady speeds are shown in Figure 30. Also plotted are the results of several human response tests (Ref. 43). Above 10 Hz the human body attenuates vibrations and acceptable levels increase continuously with increasing frequency. This effect is overestimated in all today's criteria, which are not stringent enough above the blade passage frequency.

The vibration exposure criterion, shown also in Figure 30, is a general guide for the evaluation of human exposure to whole-body vibration which has been developed and agreed upon as International Standard (Ref. 44) and which applies to such typical tasks as flying an airplane or helicopter. The times listed in the criterion are concerned with the preservation of working efficiency. The limits are called fatigue or decreased proficiency boundaries.

A critical evaluation of today's helicopters shows that vibration levels are still too high, exceeding in some cases considerably the threshold of discomfort and the fatigue boundary. This situation yield increased pilot workload. In addition, it has to be considered that the NOE flight spectrum includes substantial time periods in the low speed regime, transition flights, and rapid manoeuvering producing in some cases considerably increased vibration levels.

4. SYSTEM EVALUATION APPROACHES

With regard to the demands of the missions and of specific mission elements the flight mechanics research has the objectives to establish flying qualities criteria, which are a basis for communication between a procuring agency and a contractor in all phases of a helicopter development. Influences of all loop elements with contribution to the resulting mission performance have to be taken into account. Consequently, a formulation of the criteria in an engineering terminology is worthwhile. The primary objective of flying qualities criteria is to establish methods and evaluation levels for:

- the formulation of requirements for the rotary-wing system including baseline helicopter, visual aids, control systems etc. derived from the mission requirements,
- the guideline for the design of the helicopter system, and
- the helicopter certification.

Fundamentally flying qualities can be approached in two different ways:

- criteria for the evaluation of task performance and
- handling qualities criteria.

The advantage of an evaluation by a task performance approach is based on the derivation of flight test tasks from mission elements and accordingly, on a close relation of task performance to defined mission performance boundaries which directly can be checked in the helicopter certification. The mission-oriented formulation yields high flexibility to match new established missions and implementation of improved technology. Presupposition of the task perfomance approach is an identification or - more extensive -a standardization of the used flight test tasks representative for the mission. Deficiencies may result in the design phase when the flight vehicle or sufficient valid models of the flight dynamics do not exist.

In contrast, handling qualities criteria are more addressed to an application during all phases of helicopter development. Their affinity to the mission or mission elements has to be taken into consideration. The criteria describe the required system characteristics in degrading handling qualities levels. The main disadvantage is the high expense for their establishment, because the criteria have necessarily to be formulated in a comprehensive and generally accepted way. Due to new missions or technology, there is the risk, that the validity of these criteria have to be verified and updated.

when mental effort is also included. Nevertheless, rating scales that are accepted as being non-linear often have been used as being linear. Researchers have calculated means and standard deviations of this type of rating, though caution should be exercised when subjecting data like these to any analytical process (Ref. 40).

Among several rating scales the Cooper-Harper (Ref. 1) scale for handling qualities meets the requirement to be a widely accepted method. It was very carefully developed, and test pilots are experienced to using it. The scale is reproduced in Table 8, and the following points should be noted about the scale and the method of using it (Ref. 40):

- The intention of the scale is to obtain an absolute assessment rather than a pure comparison with other vehicles or configurations under assessment. Nevertheless rating scales are subjective in nature and therefore are scales of comparison based on the pilot's empirical knowledge.

- The pilot is drawn towards the eventual rating through a step-by-step process. The value judgements that he makes are presented as a series of decisions.

- The scale is aimed towards the practical application of the vehicle under assessment. The pilot's judgements are all made in the context of the defined task or mission. Therefore it is required, that the task or mission is clearly defined, and in a way that is acceptable to the evaluation pilots.

- The rating does not provide a complete assessment. It gives a shorthand guide to the quality of the vehicle, but the pilot should also why he arrived at the rating and what improvements he thinks are necessary.

- The scale is very practicable and easy to use and so it is suitable also for real flight conditions.

- The Cooper-Harper scale uses workload in a very specific but carefully defined and limited manner.

The Cooper-Harper rating scale was sometimes criticized because of its non-linearity, but the construction of a practical scale of demonstrated linearity has not yet been achieved. An additional point of criticism was that the pilot has to judge three different variables, aircraft characteristics, demands on the pilot, and taks performance at the same time and with a single rating. In order to obain more detailed informations on the interdependence of the three variables it was proposed (Ref. 41) to modify the scale, judging only one variable with one rating. This method seems to be suitable in order to obtain additional informations but it will not supersede the original Cooper-Harper rating scale being widely accepted and used for handling qualities evaluation.

In addition to the pilot rating a questionnaire and pilot report is required for a complete assessment. The answers to specific questions and the pilot's comments are very helpful for the research engineer, providing more insight in the evaluation process and showing possible short-comings of the experiment design. Therefore, this questionnaire has to be prepared very carefully and in close connexion with the specific experiment.

For any test program involving human pilots, and especially for an experiment in which subjective assessments are being made, it is vital to ensure that the subject pilots will give valid results. This simple requirement is not easy to fulfill because normally pilots are not representative of the community at large. In general, trained test pilots are likely to be the most suitable persons. They often have a wide experience of different types of aircraft, they have been specially trained in making assessments, and they usually looking critically at aircraft systems. In addition the experienced test pilot will have taken part in many experiments, and so he will be able to help the researcher to set up a good experiment.

For specific test programs it may be appropriate to choose subject pilots representing a mixture of test pilots and operational pilots, having different experiences and assessment skills. However, the research engineer has to be very careful in the choice of these subject pilots in order to obtain valid evaluation results.

3.5 EFFECTS OF DISTURBANCES

Because of secondary effects in the lift generation system, the propulsion system, and the control devices each aircraft shows more or less undesirable motions and effects which together with external disturbances impair and stress the pilot while he is completing his flight task. These include low frequency flight mechanical problems like control and vehicle couplings, vehicle instabilities, and gust disturbances as well as high-frequency effects like vibration and structural instabilities. Depending on the frequency and the amplitude of these disturbances the pilot either has to react by generating suitable control inputs or he will be impaired by the deterioration of the aircraft's ride qualities. Thus, with respect to critical phases of a mission, the overloading of the pilot cannot be precluded. This is true especially for helicopter pilots completing current military missions.

One of the most difficult problems that the designer must solve in each new rotorcraft development program is the vibration problem. The inherent tendency of a rotary-wing lifting system to generate periodic forces at blade passage frequency and its multi-

in accomplishing a given task by a mathematical model. During that period the human has been characterized as a servo-compensator, a sample data controller, a finite-state machine, an optimal controller, and most recently as an intelligent system (Ref. 36). Currently there is no clear consensus about the utility of available model-based methods for assessing pilot behaviour and performance in realistic helicopter missions. Nevertheless, the use of mathematical modelling as a tool for analyzing the human pilot offers great potential benefits and therefore the development of such models is of substantial interest to researchers.

Flying an aircraft imposes a load on the pilot who has to expend an amount of physical and mental efforts to accomplish the task. This simple statement points to the difficulty of defining pilot workload and a review of the literature highlights the diversity of interpretation and the existing vagueness (Ref. 37). There is no one generally accepted definition but several authors have identified effort as the main theme in their concept of workload. The idea of workload as effort is one with which many pilots would agree. It takes into consideration the individual ways in which pilots respond to the demands of the flight task by allowing for variables as natural ability, training, experience, age and fitness. However, there are other important aspects of the flight task which may be considered to be equally relevant in forming concepts of workload.

In their scale, Cooper and Harper (Ref. 1) refer to "pilot compensation" using the term to indicate that the pilot must increase his workload to improve aircraft performance. They also state that "it is the measure of additional pilot effort and attention required to maintain a given level of performance in the face of less favourable or deficient characteristics". The idea that a pilot has the ability to compensate implies that he has spare capacity. This notion is also supported by other authors (Ref. 38).

Although there are many different definitions and concepts of pilot workload it is generally acknowledged that there are two main areas for consideration, they are task-related and pilot-related aspects. It may be useful to consider workload as a multifaceted concept (Ref. 37), primary facets being formed by the three variables: demands of the flight task, pilot effort, and results. Minor or secondary facets can then be formed by the various methods used for assessing levels of workload. These will be largely dependent on the experience, discipline and interest of the investigator. It follows that any reference to pilot workload must identify the particular interpretation and the method used to assess levels.

It is customary to divide workload into physical and non-physical or mental components, though it is not always easy to identify a clear dividing line between them. Cooper and Harper (Ref. 1) distinguished between physical and mental effort and gave the following definitions:

- Workload: The integrated physical and mental effort required to perform a specified piloting task.

  - Physical: The effort expended by the pilot in moving or imposing forces on the controls during a specified piloting task.

  - Mental: Mental workload is at present not amenable to quantitative analysis by other than pilot evaluation, or indirect methods using physical workload (input) and the task performance measurements. An example would be the improvement associated with flight-director type displays which reduce the mental compensation normally required of the pilot.

Ideally, assessment or measurement of pilot workload should be objective and result in absolute values. At present this is not possible nor is there any evidence that this ideal will be realized in the foreseeable future. It is also unfortunate that the human pilot cannot be measured with the same degree of precision as can be measured mechanical and electronical functions.

Methods used for assessing workload can be broadly divided into subjective, physiological, objective, and engineering techniques. The practical application of these techniques to the three conceptual areas: Flight task, pilot effort, and performance, results in a measure of workload which will have a specific interpretation depending on the particular technique selected. For assessing the handling qualities of an aircraft and the total workload in determining its suitability for an intended mission, subjective pilot rating is still the common method (Ref. 39). It is likely that this will continue to be the case for the foreseeable future.

In this situation there is a definite need for a standarized approach, so that a generally accepted method for subjective assessment can be developed and adopted. Even if it cannot be agreed that the method is optimal, application of a single method should bring considerable benefits. Pilot assessment of workload could then be properly influential, not only as a measure to back up other methods, but also as a primary measure in its own justification.

The use of rating scales results in the allocation of a numerical value to the quantity that is being measured. Not unnaturally, researchers wish to use statistical and mathematical procedures on the numbers so obtained, and so most of the rating scales that have been devised have been intended to be linear. Although rating scales have shown good correlation with objective measurements in purely physical tasks, there is no reason to expect that any of these scales should be linear with respect to any physical variable

Voice Technology

At present there is an extensive interest in voice technology for advanced helicopter cockpits although the auditory sense may also be close to saturation with communications and normal subliminal monitoring of engine and rotor noise (Ref. 21). Nevertheless, interactive voice may be an effective means to relieve the visual information input and the manual output. For application of voice technology in aircraft two aspects have to be considered (Ref. 30): voice output and voice recognition. Recent research programs have shown that in both areas the technology does exist now for the use in the cockpit environment even in the helicopter's noise and vibration environment. The basic assumptions for this statement are: (1) a voice output device with a demonstrated intelligibility at least as good as current intercommunication systems; (2) the availability of a speech recognizer with 100 word vocabulary with the capability of training by two users and having a demonstrated accuracy of 95 to 99.9 percent under all flight conditions. Under these assumptions various applications are under consideration, as listed in Table 7.

In general, voice technology should be utilized only if it aids the pilot in accomplishing his tasks. Advocates of this technology must consider the following parameters of the cockpit environment: (1) physical/emotional stress of the pilot; (2) multicommunication inputs/outputs; and (3) the multiplicity of tasks to be performed in a limited time-frame.

Side-Stick Controller

Conventional cockpit controller configurations provide substantial disadvantages especially in missions including dynamic manoeuvering like NOE flight. It was shown that flying these tasks with increased application of collective control, providing a direct force control, may increase the task performance (Ref. 31). Conventional controller configurations are inadequate for such control strategies due to high control couplings and problems in the dynamic use of collective controllers. In order to avoid those and additional problems of conventional controllers and in order to free a hand of the pilot for mission management tasks new controller configurations have been considered and developed under the assumption of a fly-by-wire/light control system installed in the helicopter.

Various prototypes of side-stick and side-arm controllers were evaluated in ground simulation and flight test programs with the controller configuration, the level of stability and control augmentation, and the mission demands being the primary experimental variables (Refs. 32, 33, 34). In Figure 28 different controller configurations which have been under consideration are shown. The studies demonstrated that a four-axis controller with small deflection in all axes was prefered by the pilots against both a four-axis stiff-stick design and a design having limited deflection in only the pitch and roll axes. Because multi-axis side-stick or side-arm controllers will be part of very sophisticated control systems incorporating electrical or optical signal transmission and mission optimized control laws, the full assessment of the controller configurations must include this total concept.

3.4 PILOT BEHAVIOUR, WORKLOAD

The thesis that the pilot is both pivotal to mission success and, as a consequence, acts as a ultimate constraint on mission performance, is now generally accepted. However, this human component of the flying system continues to be pushed closer to his physiological and psychological limits. The design of the aircraft and aircraft components can be changed, but the basic human capabilities and limitations cannot be changed. It is pointless to build aircraft with superb performance and to man them with highly intelligent, highly trained pilots if restrictions on the rate of information flow from the machine to the man and on the rate at which the man can make inputs to the machine are the limiting factors in the performance of the overall man-machine systems (Ref. 21). In this situation it seems to be helpful to review very briefly the man's capabilities (Ref. 35).

It is estimated that 80% of flight information is visually acquired. Though there are limits to man's acuity and thresholds, the spectrum of detectable energy, range of encoding possibilities, and capacity for multiple inputs far exceed that of any other information channel. The visual channel is reasonably resistant to noise in the typical cockpit environment.

Most of the remaining 20% of the pilot's flight information is acquired auditorily. Speech is a principal mode. Audition is moderately succeptible to noise and requires moderately high energy levels in order to ensure that all information is transmitted. While not completely a single channel system, audition is significantly more limited than vision to multiple inputs.

Kinesthesis/spatial orientation account for a minimal amount of the pilot's flight information, largely in terms of a background flow of relatively gross data which remain unobtrusive until it deviates from the expected.

The data processing capabilities are the significant arena of pilot overload, being characterized by a narrow range of information capacity. These limitations are considerably augmented, however, by his capacity to predict, extrapolate, and develop unique solutions.

For over 30 years several attempts have been initiated to describe the human pilot

ture, giving the designer much more flexibility with respect to the solution of different problems. So, the question "What can be realized?" will be more and more superseded by the question "What is necessary to realize for a specific task or mission?" (Ref. 20).

3.3 MAN-MACHINE INTERFACE

The human input channels for any information is limited to the five senses of seeing, hearing, feeling, tasting, and smelling. Up to now the senses of tasting and smelling are unloaded during flight operations. The pilot's sense of feel is partially utilized only for the seat-of-the-pants cues for flying. The senses of seeing and hearing are used by far most extensively for gathering information while flying an aircraft. The pilot's outputs are normally limited to limb and head motion and speech (Ref. 21).

Visual Information

In the flight environment vision is undoubtedly the most important input sensor for the pilot. Visual cues are the only source of reasonably accurate positional information. The visual world surrounding the pilot at any given moment is of great complexity and includes a diversitv of meaning and importance for the perceiver. In visual meteorological conditions (VMC) providing adequate visibility and reference to external visual cues the pilot normally is able to select the information necessary for the piloting task. If flight operations at night or under adverse weather conditions are required the situation is more complex because the pilot needs additional information displayed in the cockpit.

Modern civil helicopters are designed and certificated for instrument meteorological conditions (IMC) based on airworthiness criteria for helicopter instrument flight provided by the Federal Aviation Administration (FAA) and other national civil authorities. Up to now the IMC operations are severely limited due to different factors, among those are lacking pilot informations. Figure 22 (Ref. 22) presents a typical envelope for helicopter IFR approach including the various limitations.

There is an actual and anticipated growth of helicopter operations in IMC with particular interest to exploit the helicopter's unique capability to fly at very low airspeed. This may lead to improved IFR criteria, including requirements for additional instruments like attitude director indicator and horizontal situation indicator. Figure 23 shows typical flight director displays being under investigation for improved pilot information (Ref. 23). Equivalent electronic displays (Fig. 24) are also under discussion having the objective to integrate a variety of information into one format (Ref. 24).

For military missions like nap-of-the-earth flight at night or in adverse weather conditions, the pilot must have sufficient external reference to fly safely close to the ground. Therefore, night-vision aids are required. Current conception for advanced attack or anti-tank helicopters is to superimpose an imagery from a forward-looking infrared (FLIR, sensor or a low light level TV-camera (LLL-TV) with flight symbologies and weapon-control symbologies. The combined display will be presented to the pilot on a panel-mounted display or a head-up display or a helmet-mounted display. In case of the helmet-mounted display all information is presented to the pilot on a 2.5 cm cathode ray tube covering one eye. It is expected that the pilot with his other eye takes care of additional matters like the peripheral scene and the instrument panel (Ref. 21).

In Figure 25 the display mode symbology for the Pilot Night Vision System (PNVS) of the AH-64 helicopter is shown (Refs. 25, 26). Depending on the flight condition three different flight control symbology formats are provided for enroute flight, for transition to hover, and for hover and bob-up. It is common opinion that in certain helicopter missions the sense of vision is utilized to the point of saturation. Modern display technology may enable the designer to provide any information the pilot needs even under night and adverse weather conditions; but does the designer really know what the pilot needs?

It should be supplemented here that several previous studies have shown that for constant pilot workload a tradeoff exists between control system complexity and cockpit display sophistication (Refs. 27, 28). In other words, this hypothesis (Fig. 26) says that a very advanced pilot information system could compensate for degradation of a flight control system and a very advanced flight control system would minimize the need for display sophistication. Together with the practical consideration of cost, which normally increases with sophistication, this relationship seems to be very essential for the design of future pilot information systems having in mind the complexity of modern cockpit display systems.

In connection with pilot perception, a scale was proposed (Ref. 29) providing a more fine grained quantification of available outside visual cues (OVC) than simply specifying IMC or VMC. This scale is shown in Figure 27. The scale was developed on the basis of comments by helicopter pilots operating in low visibility and allows quantification of minimum acceptable levels of controls and displays in terms of the pilot's ability to perceive aircraft attitude, position, and velocity via outside visual cues. In order to account for the effect of artificial aids implicitly, a similar scale for quantification of the usable cue environment (UCE) was developed including all available vision aids (Ref. 8).

Forward Flight, Basic	**Low altitude operations**
Straight and level	Terrain flight navigation
Climb/descent	Low level flight
Climbing/descending turns	Contour flight
Acceleration/deceleration	NOE-flight
Hover, Basic	Bob up, down
Takeoff to hover	Sidestep
Hover	Lateral dash/quickstop
Hover checks	Lateral jinking
Hover turns	External load placement
Forward Hover	**Weapon Delivery**
Hover out of ground effect	Hover fire
Takeoff	Running fire
Normal	Air-to-air
Maximum performance	**Air Combat Maneuvering-Offense**
Short field	High yo-yo
Obstacle clearance	Low yo-yo
Terrain flight	Horizontal scissors
Confined area	Side flare quick stop
Cross slope	Wingover attack
Up slope, down slope	Barrel roll attack
Approach and Landing	**Air Combat Maneuvering-Defense**
Traffic pattern	High yo-yo defense, close range
Steep	High yo-yo defense, long range
Go-around	Low yo-yo defense, long range
Short field	Horizontal scissors, defense
Obstacle clearance	Side flare quickstop defense
Terrain flight	**Anti Submarine**
Cross slope	Ship takeoff
Up slope, down slope	Precise hover (6 min)
Instrument Flight	Torpedo delivery
Takeoff	Ship approach
Level flight	Ship landing
Turns	
Timed turns	
Climbs/descents	
Climbing/descending turns	
Acceleration/deceleration	
Autorotation	
Instrument navigation	
Holding	
Unusual attitude recovery	
Instrument approach	

Table 1. Typical mission elements

Acceleration Response Permitted	Rate Response Required	Attitude Response Required
Hover (M)	Air Refueling	NOE (A) Low Speed/Hover
⋮	⋮	⋮
Autorotation	Air Retrieval	Weapon Delivery (A)
⋮	⋮	⋮
Formation (M)	Shipboard Recovery (M)	Shipboard Recovery (A)
⋮	⋮	⋮
Departures	NOE (M) Fwd Speed	TBD
⋮	⋮	
Approaches	TBD	
⋮		
Up and Away	(M): MODERATE	
⋮		
Evasive Maneuvers	(A): AGRESSIVE	
⋮		
Weapon Delivery (M)		
⋮		
TBD		

Table 2. Required rotorcraft response characteristics as a function of mission elements
(ideal outside visual cues; moderate level of turbulence)

	Upgraded Response-Type Required Due to Degraded UCE			
	UCE = 2	UCE = 3	UCE = 4	UCE = 5
When acceleration response is allowed by Table 2	Acceleration	Rate	Attitude	TRC
When rate response is required by Table 2	Rate	Attitude	Attitude	TRC
When attitude response is required by Table 2	Attitude	Attitude	TRC	TRC

NOTES:

- UCE is OVC with the addition of all available artificial vision aids, see Figure 27
- TRC is an abbreviation for translational rate command.

Table 3. Required upgraded response type in the presence of degraded UCE (useable cue environment)

CHARACTERISTICS	CHARACTERIZATIONS	
Maneuvering Required M	Rapid 1	Gradual 0
Precise Flight Path or Space Position Control Required P	Yes 1	No 0
Target Tracking Required T	Yes 1	No 0

Flight Phase Categories are defined as the following combinations of the characterizations of the characteristics.

M	P	T	Examples
1	1	1	Ground Attack
1	1	0	Terrain Avoidance, NOE
1	0	1	Air-Air Combat With Missiles
1	0	0	Missile Avoidance
0	1	1	Hover Bob-Up & Target Acquisition
0	1	0	External Load Placement
0	0	1	Missile Launch
0	0	0	Loiter

Table 4. Categorization of flight phases

REQUIRED OPERATIONAL CAPABILITY

External Visual Conditions in Which Operational Capability is Required	Only When Position and Velocity Cues Are Available	Even When Position and Velocity Cues are Not Available
Only when Angular Orientation Cues are Available	Class I	Class II
Even when Angular Orientation Cues are Not Available	Class III	Class IV

Table 5. Classification of required operational capability

AS-365 Dauphin (US Coast Guard)

- Mechanical control linkages
- SAS duplex, 4-axis
- Automatic trim, force feel system
- Autopilot for attitude, heading, and flight director modes

UH-60A Black Hawk

- Mechanical control linkages, horizontal stabilizer fly-by-wire
- SAS duplex, 3-axis
- Artificial stick position stability, force feel system
- Autopilot for attitude, heading, speed, flight path
- Automatic horizontal stabilizer control

AH-64 Apache

- Mechanical control linkages with fly-by-wire back-up
- SCAS duplex, 3-axis
- Automatic horizontal stabilizer control
- Force feel system
- Autopilot for attitude, heading

Table 6. Control systems for modern helicopters

SYSTEM	INPUT	OUTPUT
COMMUNICATION	TUNING RADIO SELECTION	DIGITAL MESSAGE RECONSTRUCTION
NAVIGATION	POSITION REQUEST STEERING REQUEST MAP INFORMATION REQUEST POSITION UPDATE	POSITION REPORT STEERING INFORMATION MAP INFORMATION
FLIGHT CONTROLS	FUNCTION SELECTION ACTION COMMAND	FEEDBACK
SUBSYSTEMS	POWER INFORMATION REQRS. SELECT CONTINGENCY POWER REQUEST FUEL STATUS LIGHTING CONTROL SELECT DISPLAY MODE	POWER INFORMATION FUEL STATUS
CAUTION, WARNING, ADVISORY	PRIORITY SELECTION	PRIMARY ALERTING SYSTEM

Table 7. Cockpit voice applications

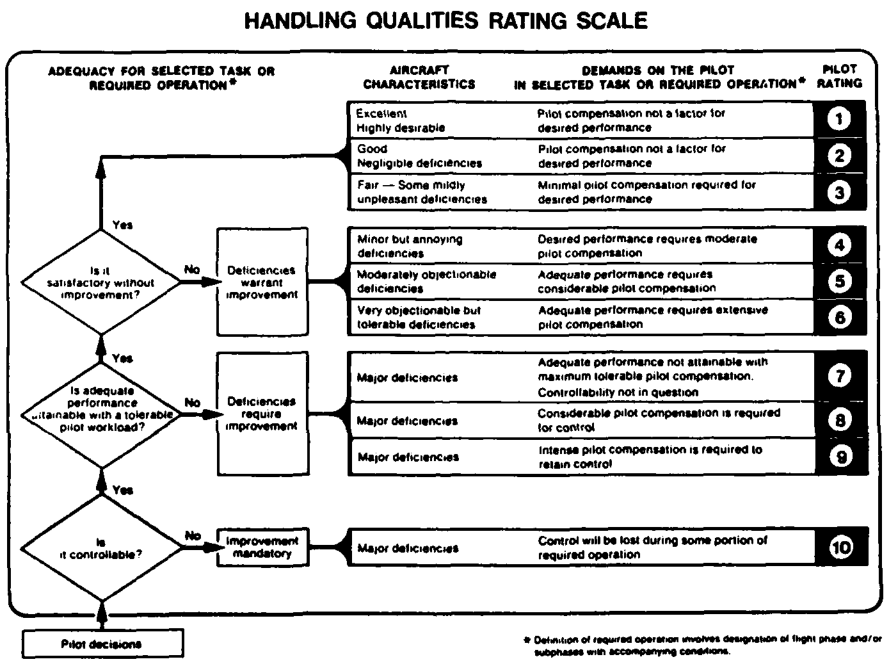


Table 8. Cooper-Harper scale

NASA/US Army Labs

- In-flight simulators
  UH-1 V/STOLAND
  Ames CH-47
- Ground simulators
  Vertical motion simulator
  Flight simulator for advanced aircraft

CALSPAN

- In-flight simulator
  X-22 A

NAE

- In-flight simulator
  NRC Bell 205

RAE

- Ground simulator
  6 DOF moving base simulator

DFVLR

- In-flight simulator
  BO 105-S3 ATTHeS

Table 9. Test facilities

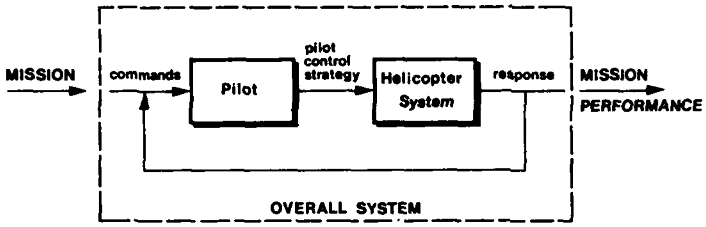


Figure 1. Block diagram of pilot-helicopter system

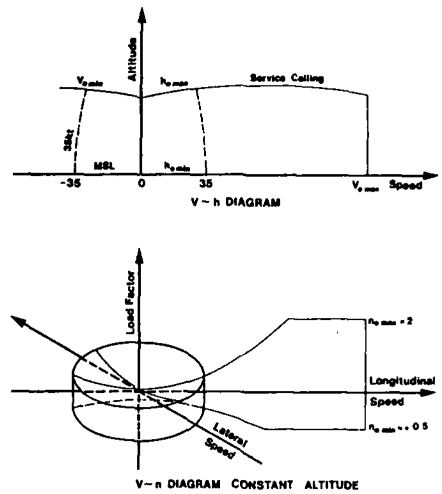


Figure 2. Typical operational flight envelopes

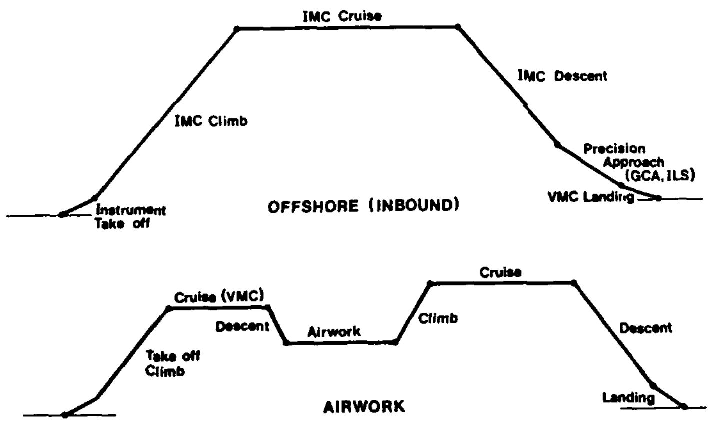


Figure 3. Flight profiles

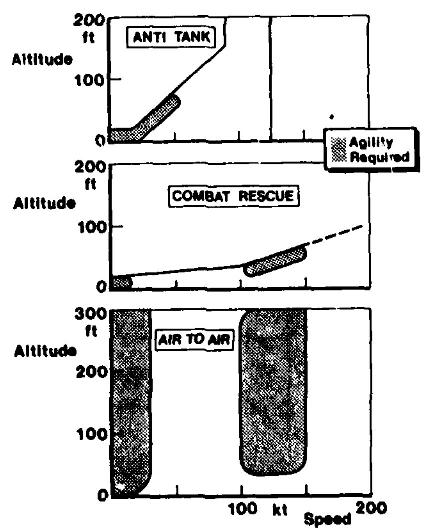


Figure 4. Demands of rotary-wing missions

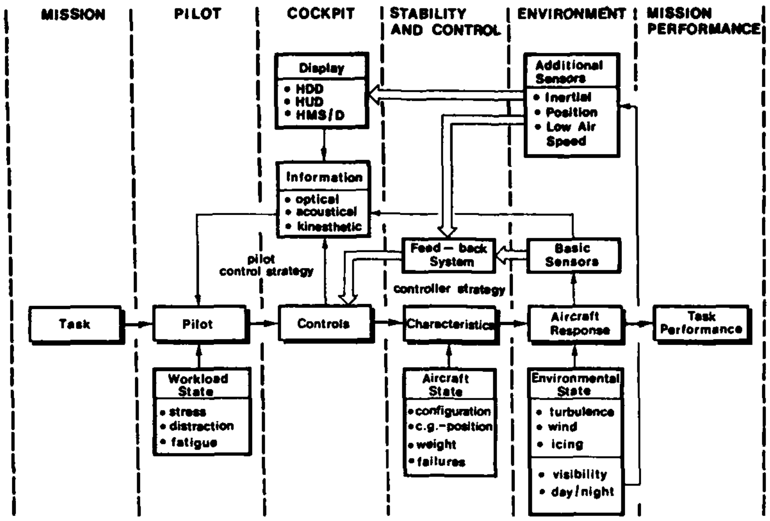
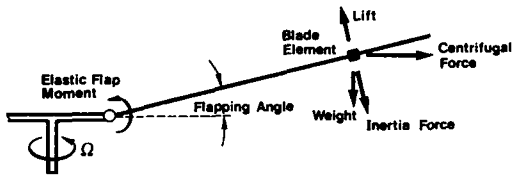


Figure 5. Factors influencing pilot-helicopter system performance

Figure 6. Forces acting on rotating blades



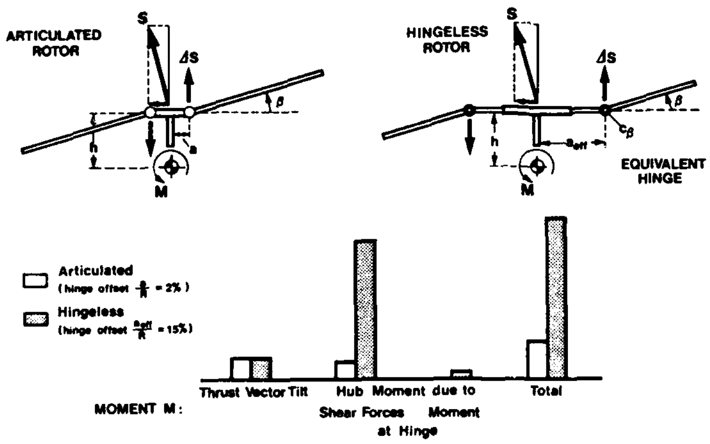


Figure 7. Rotor control moment capacity

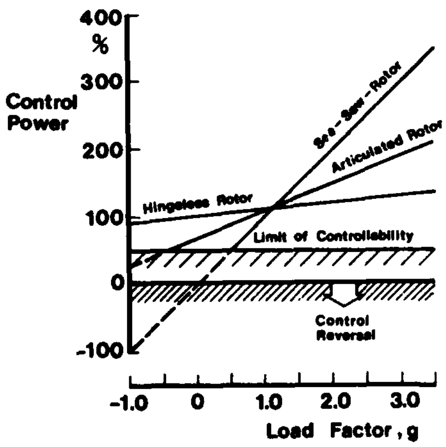


Figure 8. Manoeuver control power

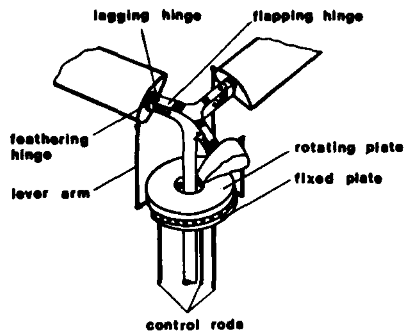


Figure 9. Swashplate assembly

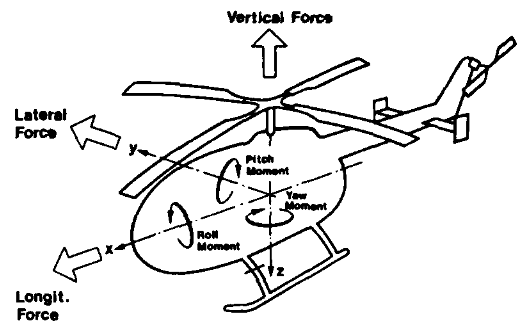


Figure 10. Body-fixed axis system

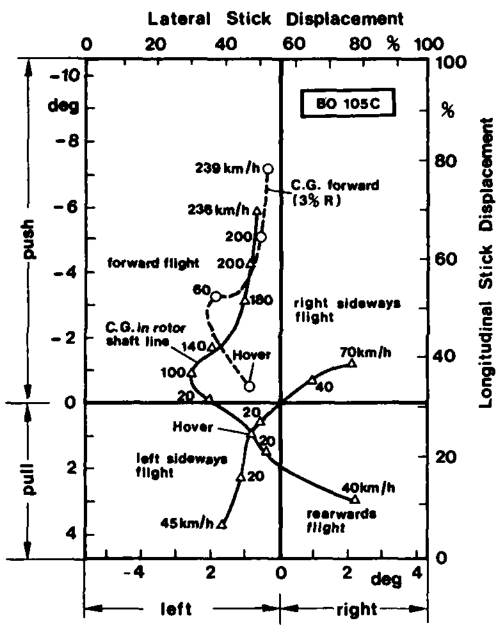


Figure 11. Rotor control requirements

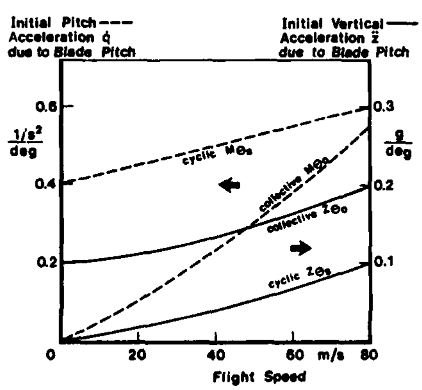


Figure 12. Helicopter control coupling

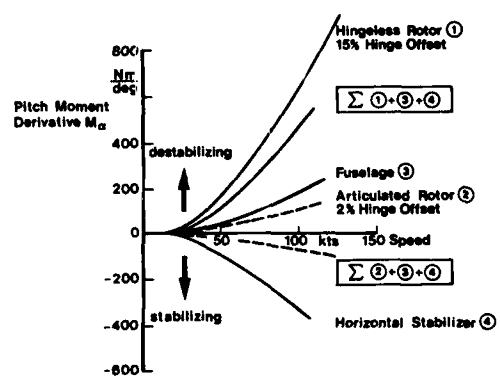


Figure 13. Angle-of-attack stability

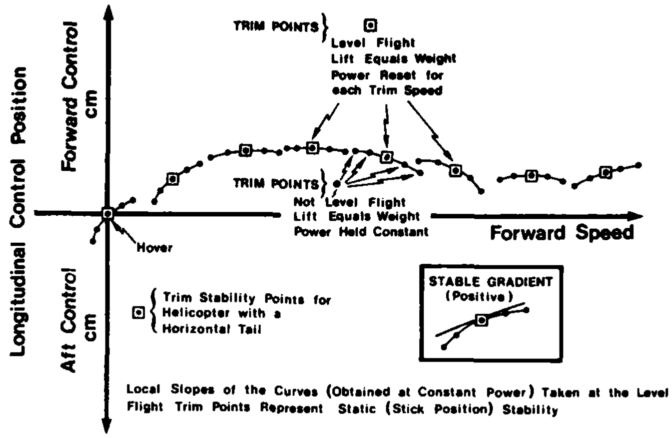


Figure 14. Static stability curves

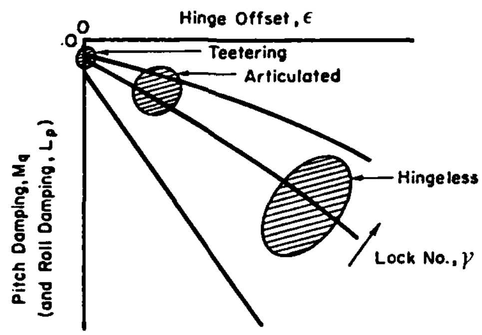


Figure 15. Pitch damping as function of rotor parameters

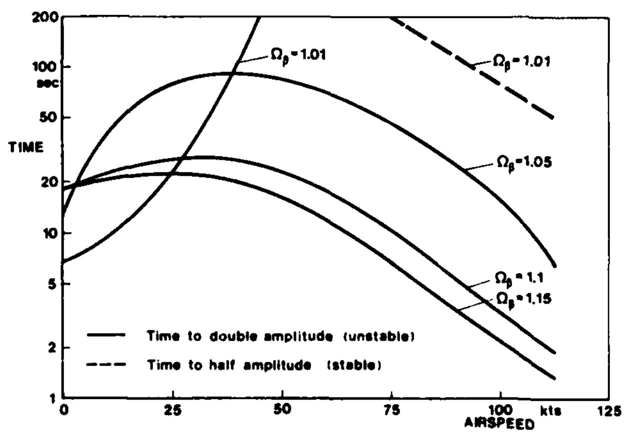


Figure 16. Helicopter dynamic stability

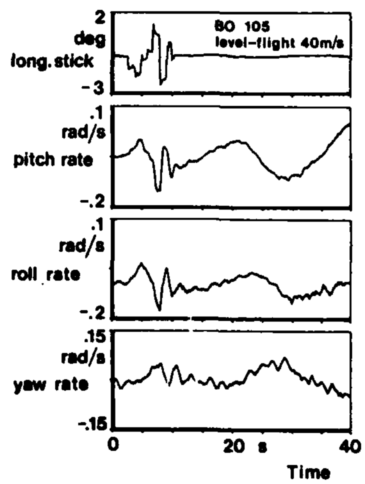


Figure 17. Inter-axis coupling

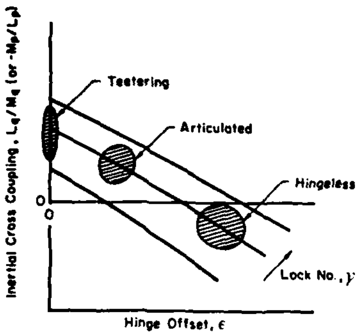


Figure 18. Cross coupling as function of rotor parameters

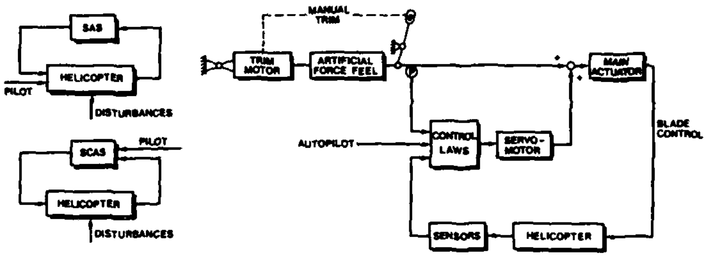


Figure 19. Stability and control augmentation system

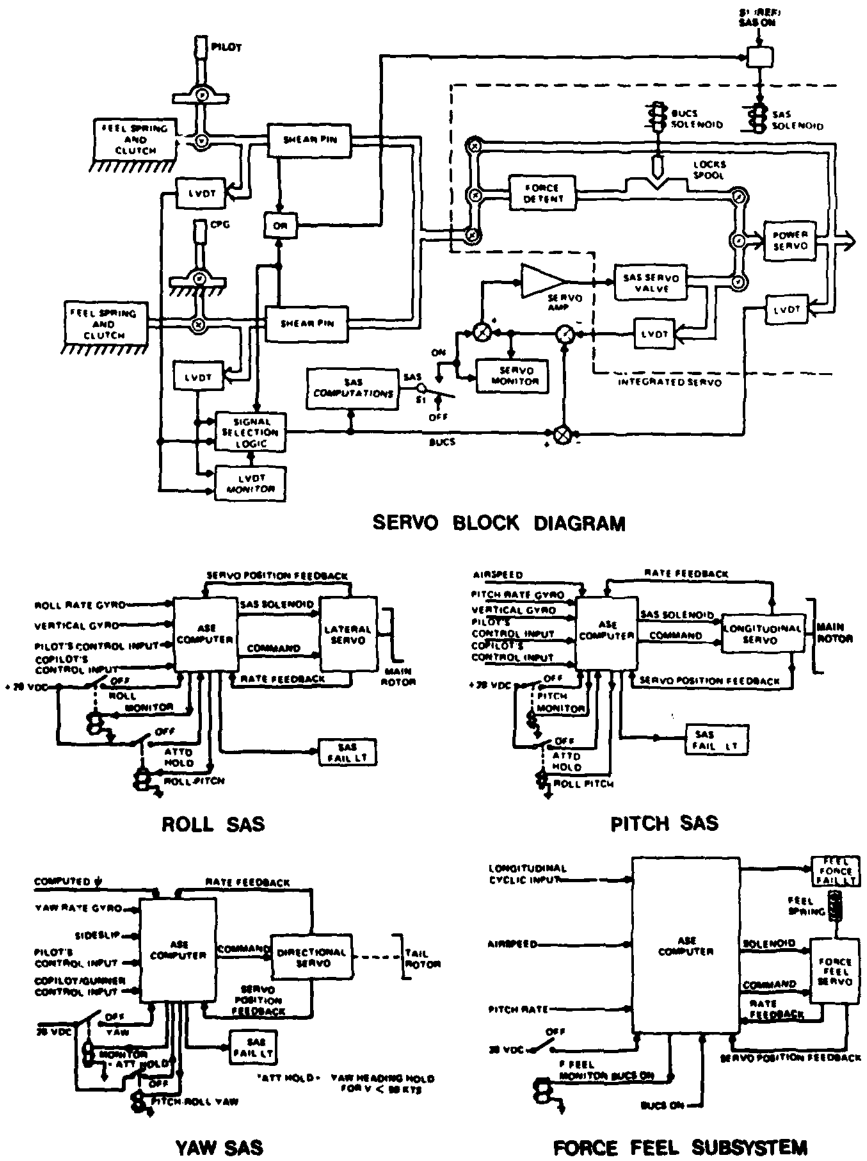


Figure 20. Automatic stabilization system (AH-64)

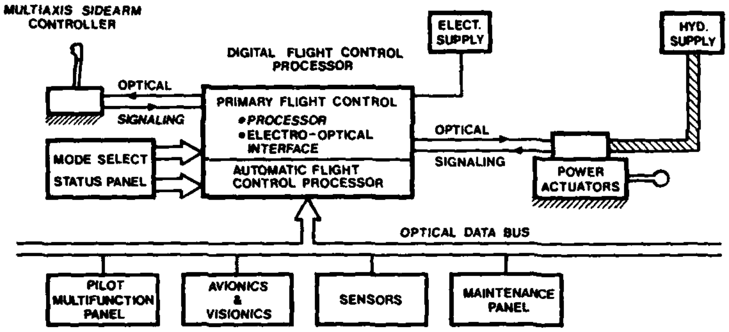


Figure 21. Advanced Digital Optical Control System (ADOCS)

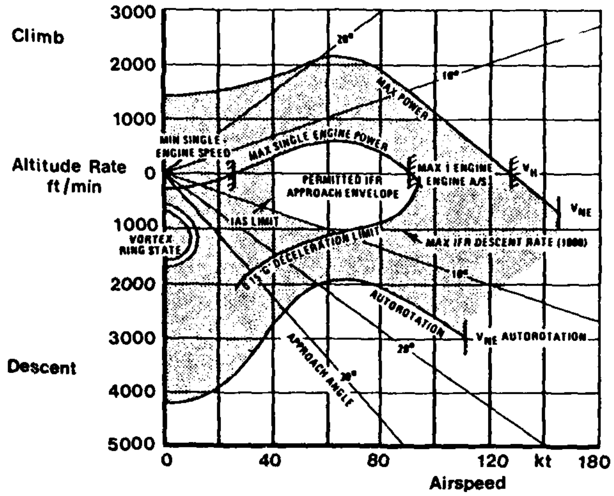


Figure 22. Typical IFR flight envelope

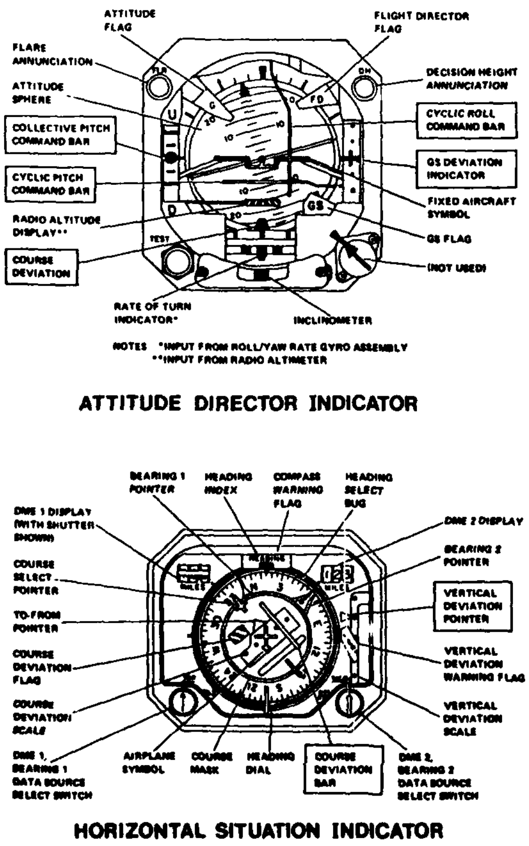
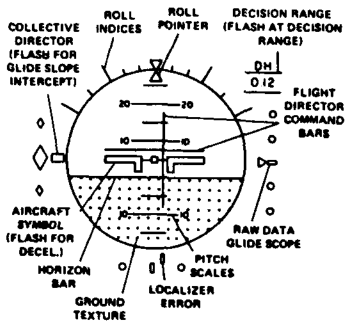
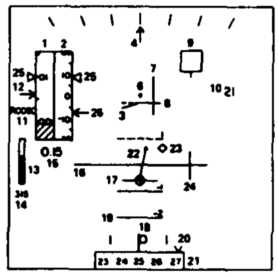


Figure 23. Flight director displays





1. ALTITUDE TAPE
2. VERTICAL SPEED
3. THRUST MAGNITUDE CONTROL DIRECTOR
4. ROLL POINTER
6. PITCH & ROLL STICK DIRECTOR INDEX
7. LATERAL STICK CONTROL DIRECTOR
8. LONGITUDINAL STICK DIRECTOR
9. LANDING PAD (APPEARS AT DECISION RANGE)
10. AIRSPEED
11. RADAR ALTITUDE
12. ALTITUDE INDEX
13. TORQUE
14. ROTOR RPM
15. RANGE
16. HORIZON BAR
17. AIRCRAFT SYMBOL (FLASH FOR DECEL)
18. SIDESLIP
19. PITCH ATTITUDE
20. WIND DIRECTION
21. HEADING SCALE
22. GROUND VELOCITY STATUS VECTOR (APPEARS AT DECEL.)
23. GROUND VELOCITY VECTOR COMMAND (APPEARS AT DECEL.)
24. LATERAL COURSE OFFSET
25. GLIDE SLOPE (FLASHES AT INTERCEPT)
26. IVSI

Figure 24. Electronic display formats

SYMBOL	INFORMATION
1. AIRCRAFT REFERENCE	FIXED REFERENCE FOR HORIZON LINE, VELOCITY VECTOR, HOVER POSITION, CYCLIC DIRECTOR, AND FIRE CONTROL SYMBOLS
2. HORIZON LINE (CRUISE MODE ONLY)	PITCH AND ROLL ATTITUDE WITH RESPECT TO AIRCRAFT REFERENCE (INDICATING NOSE-UP PITCH AND LEFT ROLL)
3. VELOCITY VECTOR	HORIZONTAL DOPPLER VELOCITY COMPONENTS (INDICATING FORWARD AND RIGHT DRIFT VELOCITIES)
4. HOVER POSITION	DESIGNATED HOVER POSITION WITH RESPECT TO AIRCRAFT REFERENCE SYMBOL (INDICATING AIRCRAFT FORWARD AND TO RIGHT OF DESIRED HOVER POSITION)
5. CYCLIC DIRECTOR	CYCLIC STICK COMMAND WITH RESPECT TO HOVER POSITION SYMBOL (INDICATING LEFT AND AFT CYCLIC STICK REQUIRED TO RETURN TO DESIGNATED HOVER POSITION)

CENTRAL SYMBOLOGY

SYMBOL	INFORMATION
6. AIRCRAFT HEADING	MOVING TAPE INDICATION OF HEADING (INDICATING NORTH)
7. HEADING ERROR	HEADING AT TIME BOB-UP MODE SELECTED (INDICATING 030)
8. RADAR ALTITUDE	HEIGHT ABOVE GROUND LEVEL IN BOTH ANALOG AND DIGITAL FORM (INDICATING 50 ft)
9. RATE OF CLIMB	MOVING POINTER WITH FULL-SCALE DEFLECTION OF ±1,000 ft/min (INDICATING 0 ft/min)
10. LATERAL ACCELERATION	INCLINOMETER INDICATION OF SIDE FORCE
11. AIRSPEED	DIGITAL READOUT IN knots
12. TORQUE	ENGINE TORQUE IN percent

PERIPHERAL SYMBOLOGY

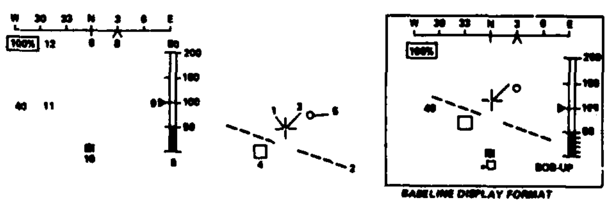


Figure 25. Pilot Night Vision System (PNVS) display mode symbology

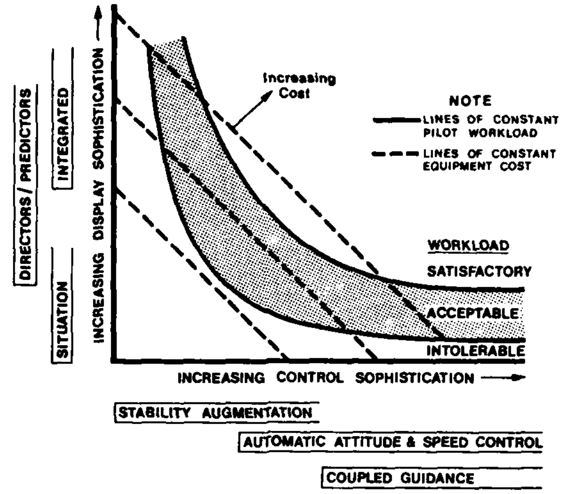


Figure 26. Hypothesis for control system/display tradeoff

	Attitude Cues	Position and Velocity Cues	OVC Level
VMC	Easily obtained.	Easily obtained.	1
VMC	Somewhat obscured. Requires full concentration to obtain continuous attitude information	Easily obtained	2
Partial IMC	Inadequate in some sectors of the visual field.	Adequate position. Marginal rate cues.	3
Partial IMC	Inadequate over most of visual field.	Position and rate cues are marginal. Rate cues are intermittently unavailable.	4
IMC	Not available.	Not available.	5

Figure 27. Quantification of outside visual cues (OVC)

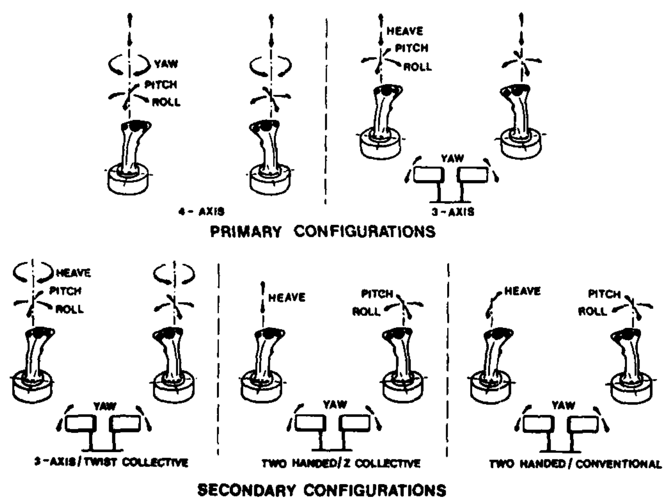


Figure 28. Side-arm controller configurations

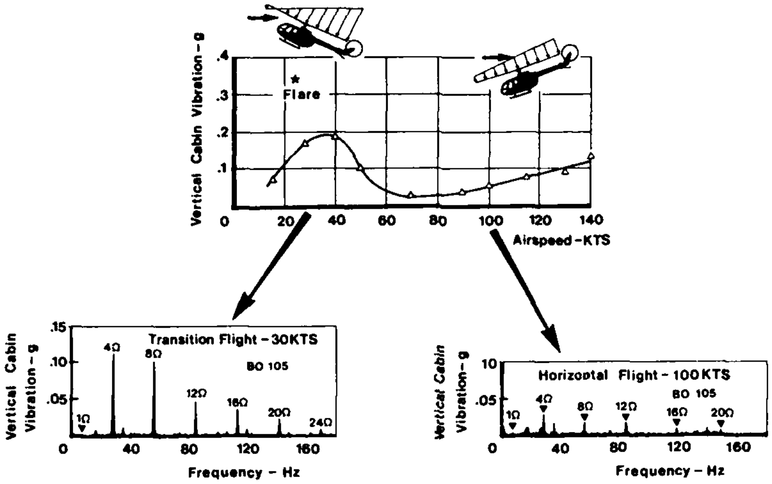


Figure 29. Helicopter vibration

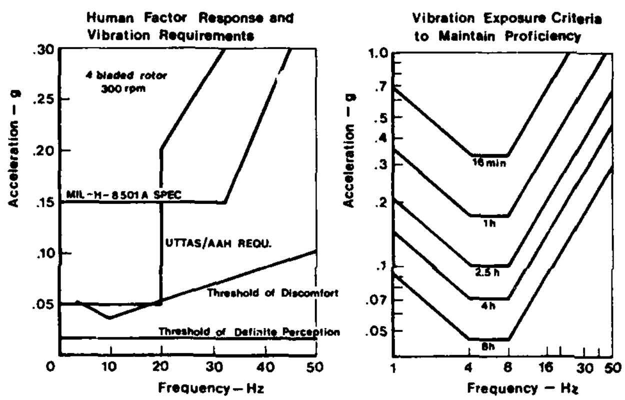


Figure 30. Vibration criteria

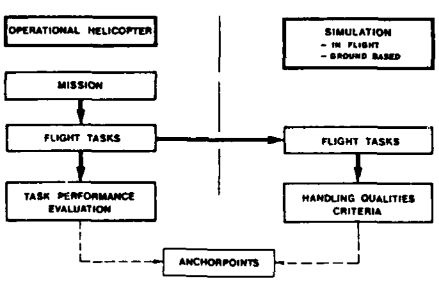


Figure 31. Use of test facilities

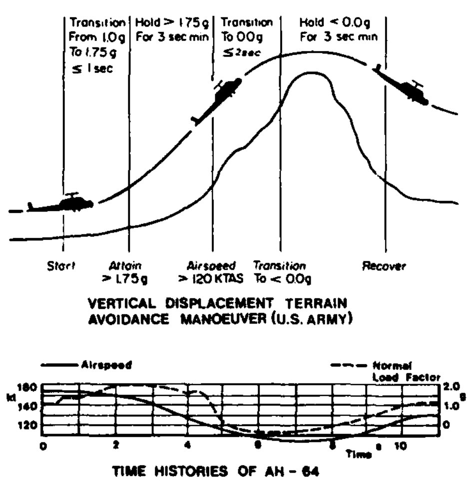


Figure 32. Terrain avoidance manoeuver

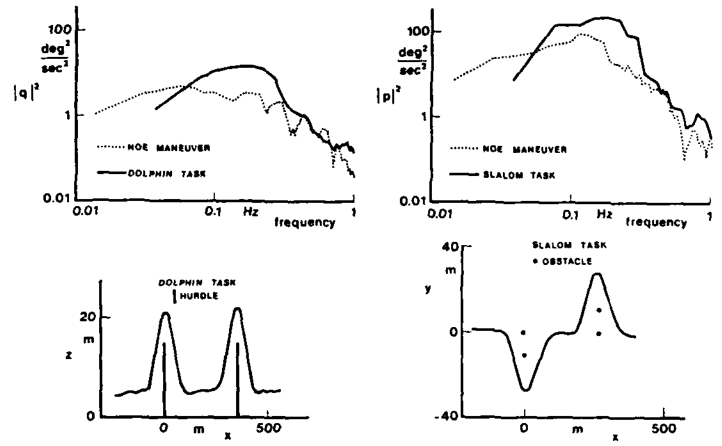


Figure 33. Dolphin and slalom flight tasks

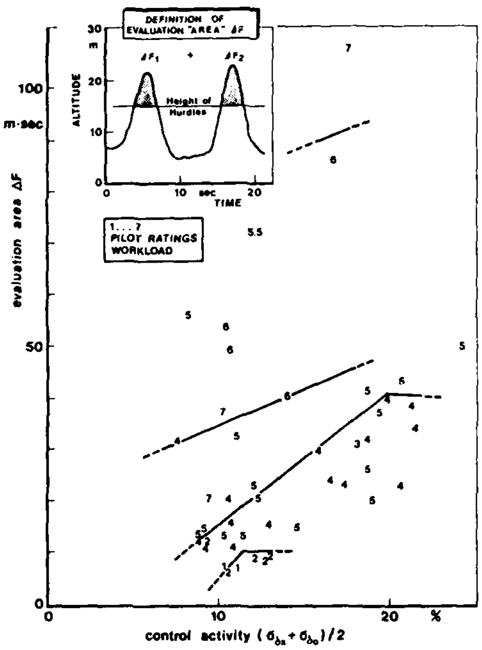


Figure 34. Dolphin evaluation diagram

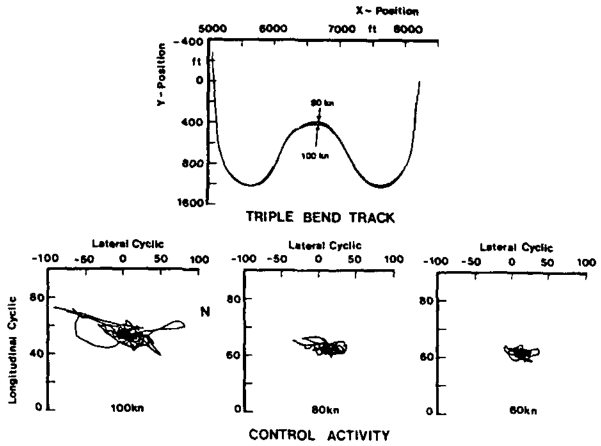


Figure 35. Triple bend performance evaluation

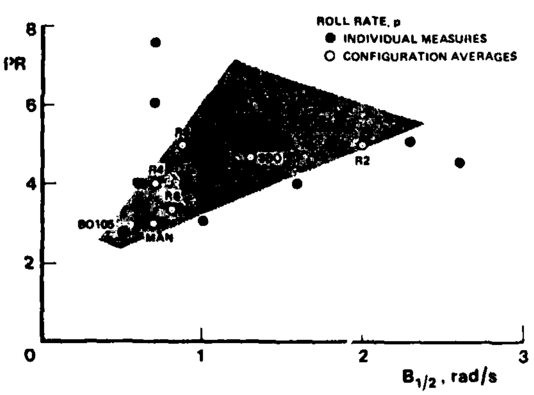


Figure 36. Roll rate half power bandwith versus pilot rating

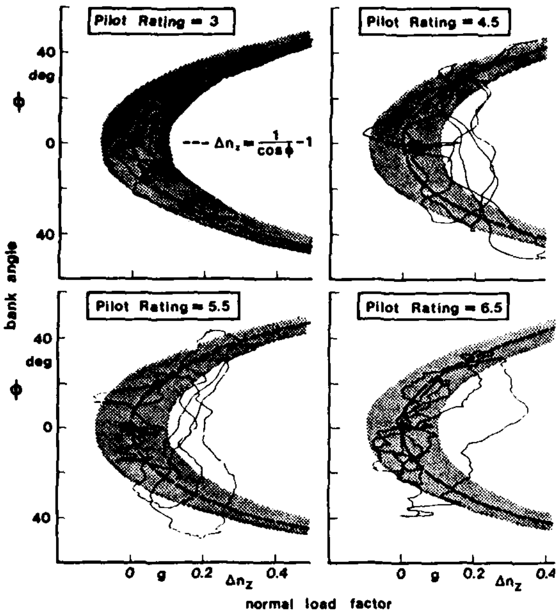


Figure 37. Slalom task performance evaluation

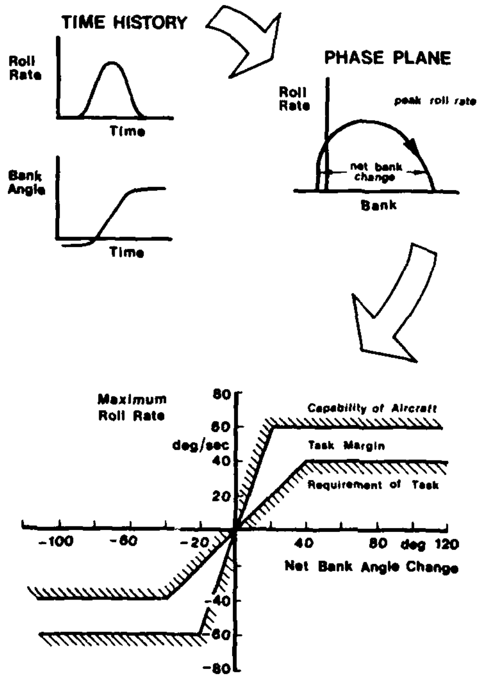


Figure 38. Interpretation of discrete manoeuver performance

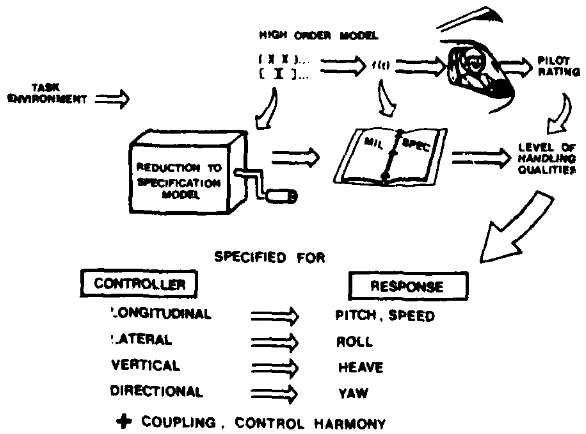


Figure 39. General approach for handling qualities specification

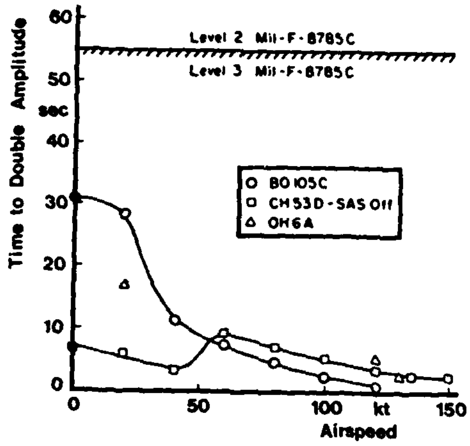


Figure 40. Comparison of fixed-wing standards with helicopter data

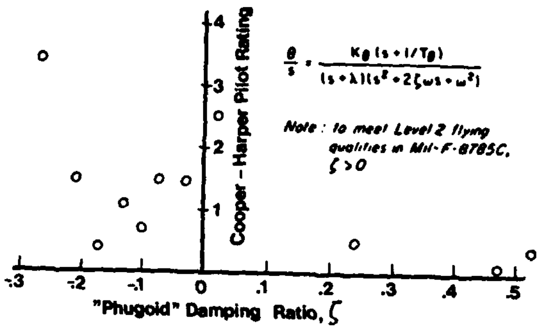


Figure 41. Pilot ratings versus damping ratio (Hover, ω<0,5 rad/sec)

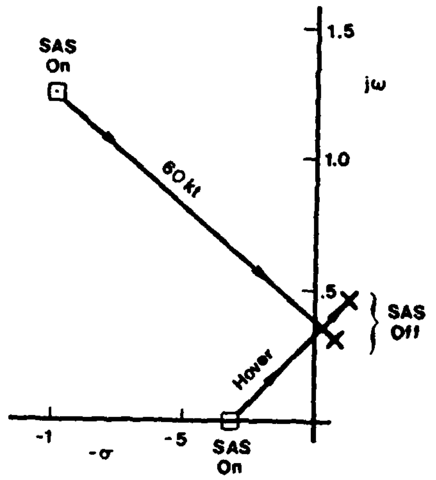


Figure 42. Effect of SAS failure on key response modes (CH-53D)

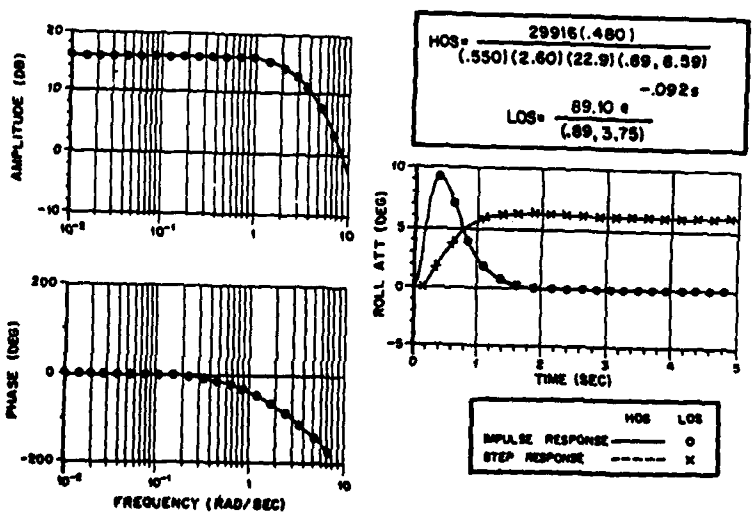


Figure 43. Roll dynamics in hover (VAK 191 B)

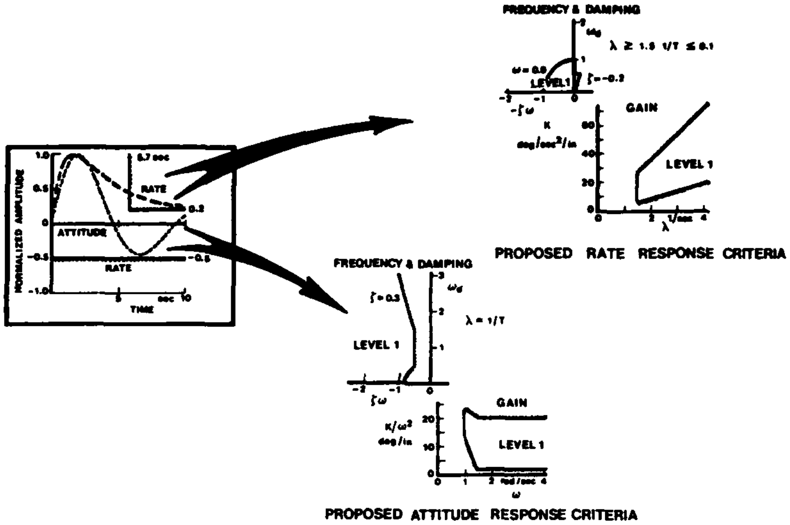


Figure 44. Proposed attitude/attitude rate response criteria

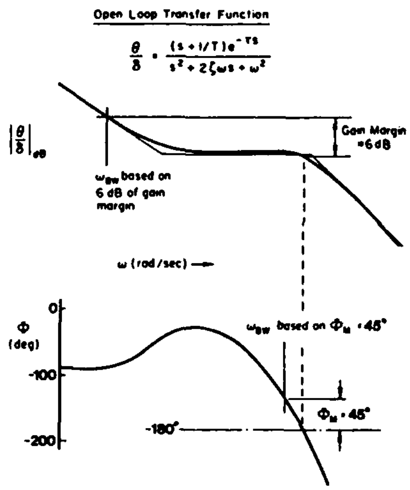


Figure 45. Bandwith definitions

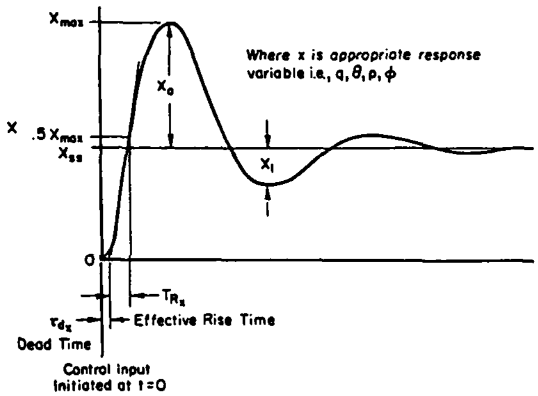


Figure 46. Definition of time domain parameters

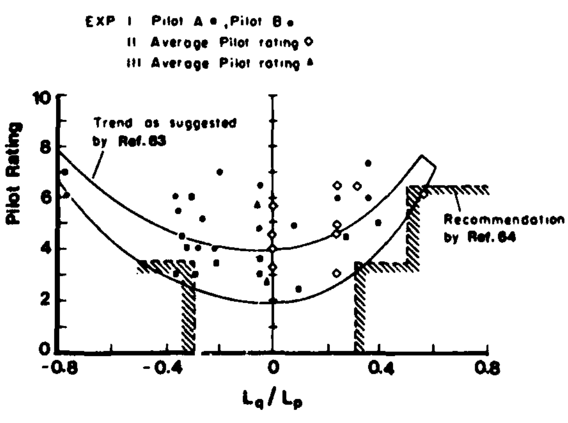


Figure 47. Pilot ratings versus pitch-to-roll coupling

THE ROLE OF SIMULATION

by

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SUMMARY

The development of helicopter flight simulation has undoubtedly made considerable progress in the past decade. Due to the helicopter specific flight regime, which is essentially characterized by low altitude flying (NOE), low to moderate speeds, and distinct stability and control behavior, the simulation of rotorcraft sets high standards to the simulation quality.

The paper gives an overview about the various simulation techniques and their specific application during research and development work. First, the details and capabilities of the ground-based simulation are discussed. Stringent requirements for real-time simulation result in the use of comprehensive math models, representing the aerodynamic and dynamic complexities of rotary-wing aircraft. In addition, great effort has to be made to simulate the environmental scenario, such as visual and motional cues. Advances are especially made on the field of generating and displaying visual imagery. The technique of computer generated images (CGI) and the progress achieved, e.g. in the field-of-view and resolution, is demonstrated.

Typical ground-based simulation tasks are described including research and exploratory investigations as well as vehicle configurational optimization, avionics, vision and weapon system integration, cockpit layout, and assessment of flight path management and mission management. The benefits of engineering simulation, such as early evaluation of system effectiveness, pilot acceptance, cost effectiveness, are discussed.

Airborne simulators and variable-stability aircraft play another important role in the aeronautical research and development. In-flight simulators are especially used in those areas where the simulation of the real world environment of laboratory simulators is limited, or when the operation of hardware components in real flight conditions is of particular importance. Details of the requirements and typical in-flight simulator objectives and tasks are presented in a state-of-the-art review.

It is concluded that modern rotorcraft simulation techniques provide a valid tool for supporting the development of future helicopter/weapon systems, including the interactions of aircraft, equipment, crew, and environmental conditions. It is demonstrated that both ground-based and airborne simulation represent an important link in the design and development of the next generation of rotorcraft systems.

1. INTRODUCTION

During the past few years, a rapidly increasing application of flight simulation during research, design and development work of helicopters has become evident. While flight simulation has been intensively used by fixed-wing designers for a much longer time, the helicopter industry has been more reserved towards simulation application during development work. There were manifold reasons for this attitude: In early ground-based simulators, rotorcraft modelling was obviously not sufficient for achieving full acceptance in development simulations. The visual display quality was not adequate to fulfill the required high standards on simulation fidelity. Deficiencies in motion cueing were further major factors to the problem.

Although variable-stability aircraft have made major contributions in the past 25 years, variable-stability helicopters have not been widely used by the rotorcraft industry as development tools for particular designs. The application of existing airborne simulators was in many cases restricted due to limitations in system actuators operations, cockpit display and navigation system capability, or in the flight envelope of the basic aircraft design.

The changing attitude towards simulator application during helicopter development work results from different reasons: Firstly, design requirements for future helicopter systems are becoming more and more demanding. It is no longer sufficient, and may become extremely costly and time-consuming, to solve problems by a classic "cut-and-try"-technique during flight testing. Secondly, there are major efforts to replace current helicopter flying qualities criteria, and to develop new specifications containing basic mission-oriented handling qualities (Ref. 1). Special considerations must be given to mission flight phases and environmental conditions (day/night, visibility, terrain nature, atmospheric conditions). Finally, similarly as with military fixed-wing aircraft, future military helicopters will represent increasingly complex weapon systems. Development of these systems must not only concentrate on the basic vehicle optimization, but has to put the main emphasis on the integration of all interrelated elements, such as basic aircraft, flight controls, displays and vision aids, weapon system and the human element, the pilot. Total system simulation techniques are required to handle the complex interrelation of the above elements and to support engineering and aircrew-aircraft integration efforts, both in research institutes and in helicopter industry.

Neglecting the pure non-real-time simulation, there are basically two types of simulation, the real-time ground-based simulation (with man-in-the-loop), and the airborne (in-flight) simulation (Figure 1). There have been numerous discussions about the specific benefits, advantages and disadvantages of these simulation techniques in the different stages of aeronautical research, design, and development work. The rapid advances in recent years in the level of sophistication and fidelity of modern ground-based simulators have sometimes

called into question the future role of in-flight simulators in research and development processes (Ref.2).

It is the aim of this lecture to demonstrate that both ground-based and airborne simulation are complementary techniques, and that both techniques will continue to serve a highly useful role and an important link in the design and development of the next generation of advanced aircraft weapon systems. Only a short description of todays's situation of engineering simulation can be given in the scope of this survey, and the complexity of the problems involved suggested better to cite competent authors rather than to generalize too broad. More profound or complete information should be gathered from the various publications that are mentioned in the context.

2. GROUND-BASED SIMULATION

2.1 HELICOPTER SIMULATION REQUIREMENTS AND REALIZATION

Mathematical Modelling / Computer Requirements. Technical simulations are based on the mathematical description of physical and mechanical processes, called the general equations of motion, which are typically highly nonlinear differential equations to be solved in closed form for only the most simple cases. Any system, subsystem, or vehicle component that produces forces and/or moments has to be considered in this math modelling, and the complex interactions and cross-coupling effects among the individual components are often much more difficult to describe.

However, the simulation of rotorcraft even goes beyond these strong demands. Rotor blade aerodynamics depend both on the radial distance from the rotor axis and on blade azimuth. They are highly influenced by unsteady aerodynamic and aeroelastic effects due to blade bending and torsional deflections. In certain flight situations, parts of the rotor disk enter nonlinear aerodynamic conditions such as stall or high MACH number flow (Ref. 3). Additional aerodynamic complexities arise from rotor-fuselage interferences or low level hover and ground effects which are difficult to investigate in flight test or in the wind tunnel.

The attempt to implement these complex relationships on a computer results in the demand for very large computer capacities and in complex computer programs that are, at least for the time being, impossible to execute in real-time. So such off-line simulations are especially used when the time parameter is of minor importance, or when the process occurs in some way periodically, as e.g. to analyze blade loads, rotor dynamics, and stability and control problems.

For real-time simulation, be it ground-based or in-flight, fundamental simplifications have to be made, but the extent to which this is tolerable depends on the actual problem. TABLE 1, developed by R.T.N. CHEN of NASA-Ames, indicates a matrix of possibilities for math models based on including different representations of the aerodynamics and rotor dynamics. Linear aerodynamics implies simplifications such as infinitely stiff rotor blades, small flapping and inflow angles, and simple strip theory with no consideration of stall effects or MACH number. With such a model, much useful work of a generic nature can be performed. However, if it desired to investigate boundaries of the flight envelope, then even in generic studies the effects of compressibility, stall and other nonlinearities must be included.

The rotorcraft simulation capability applied at MBB to meet the needs of research and development represents the essential effects of nonlinear rotor aerodynamics, rigid body and rotor dynamics and considers all other aircraft components (fuselage, wing, empennage, and tail rotor). The aerodynamic model is based on blade-element theory, including the effects of compressibility, stall and reverse flow, obtained from experimental airfoil data. Rotor downwash is modelled by a modified momentum theory, with quasi-dynamic downwash effects being simulated by introducing a time constant in the induced velocity calculation (Ref. 4).

The necessary bandwidth for modelling rotor dynamics is determined by the extent to which, for example, aeroelastic effects or high-gain feed-back control systems are to be investigated. The frequency range of interest is summarized in Figure 2 which shows the typical domain of body modes, engine and rotor modes being in close proximity. To model the rotor and body modes coupling adequately, the rotor is represented by an individual blade model, considering flapping of each of the actual blades separately (Ref. 5). A summary of the individual blade calculations is given in Figure 3. Different integration step lengths are used for the blades and body motion, which leads to the demand for a computer that is able to support parallel processing computation. As drive train frequencies can be well within the range of body and rotor modes, the influences are taken into account in a simplified first-order engine and governor dynamics model including fuel flow characteristics (Figure 4). Piloted simulation results showed that the inclusion of the rotor speed / power plant degree of freedom improved simulation fidelity substantially.

Once the simplified math model has been implemented on the computer, it is another problem to determine how well the model equations represent the actual aircraft. A first and easy step is to compare the steady state trim values of power, flight control positions and aircraft body angles for given flight conditions (airspeed, height, flight path angle...) which result from simulation and flight test results. Figure 5 demonstrates good agreement in the control correspondence, especially in and around hover flight. This flight regime is a primary target of the helicopter simulation task as it represents a flight state in which handling qualities are strongly influenced by a number of complex rotor and interference effects including rotor downwash effects contributing to lateral control trim position in low speed forward flight.

Validation of the simulation model becomes much harder when it enters the stage of comparing dynamic time histories resulting from various control inputs with those data obtained from actual flights. Due to the manifold interactions between the different rotorcraft components it is often very difficult to determine the most relevant parameter(s) and to decide how to change the math modelling to improve the desired fidelity without harming the physical fundamentals. Ref. 6 details which ambitious efforts may be taken to validate helicopter simulation using flight data. Here basic flight tests for various test maneuvers were performed, accomplished in no less than 72 flights with 123 flight hours. Subsequent parameter identification will be applied to develop a systematic and semi-automated procedure for upgrading the math model. Figure 6 illustrates the results which can be achieved after a careful model adjustment. Again the simulation and flight test time response on control inputs show good agreement.

Up to this point, we have considered only the simulation of the rotorcraft itself. Yet more simulator problems arise from the need to simulate the environmental scenario around the aircraft. A relatively easy task is the simulation of the atmospheric conditions that effect the aircraft in its flight through the boundary layer of the earth. Evidently there are no helicopter-specific requirements so far, so we can profit from the experiences in fixed-wing simulation. State-of-the-art math models for normal atmosphere (temperature, atmospheric pressure and density, speed of sound as a function of height) as well as wind, wind shear and turbulence models, both spatial or time-dependent, are available.

It is well understood that the pilot's flying qualities evaluation is predominantly based on the impressions he gets from the stimulation of his sensual receptors, i.e. visual, motional and auditorial cues. For a long time, the requirements for sufficiently providing the pilot motion and visual cues were, despite their immense costs, the most restricting factors for running rotorcraft simulation programs in real-time, but with the advent of a new generation of most powerful high-speed and high-capacity digital computers these problems are diminishing.

Motion systems. Motion sensations to the human body are always created by accelerations and are received directly by various physiological receptors. Motion information is processed instinctively with no reaction time-lag caused by attention, identification or interpretation. Hence perception of acceleration means that at least 90° phase lead, and often 180°, is obtained relative to visual observation of the world (Ref. 7). Due to this early indication of change in aircraft state, motion is often considered a more important simulator cue than visual information. Evaluating motion versus visual cues in piloted flight simulation, Ref. 8 concludes that it is not always sufficient for the pilot to achieve a similar performance in the simulator as in flight; it is also necessary that he should adopt the same control strategy.

Unfortunately there exists no obvious and accepted measure of motion cue requirements, but it is generally agreed that motion simulation is required to obtain full potential pilot performance (e.g., when an unstable or near neutrally stable aircraft is to be simulated). As stated in Ref. 9, in a more specific sense, motion simulation is required:

(1) when expected motions are above human sensory or indifference thresholds;

(2) when expected motions are within the sensory frequency range, that is, above 0.2 - 0.5 rad/sec;

(3) if full pilot performance (e.g., tracking) is desired;

(4) when a degree of face validity or realism is required to gain pilot acceptance of the total simulation.

TABLE 2 lists some motion platform requirements for critical terrain flight maneuvers that resulted from fixed-base simulations of NOE flight operations. The requirement is for all axes to produce these quantities simultaneously. These and other requirements such as rotational and translational motion platform thresholds are given in Ref. 9.

No widely accepted methodology for motion simulation is at hand. However, some concepts have proven to be suitable to produce motion sensations in the simulator that are close to those in flight. Special washout filter techniques take advantage of the washout characteristics of human motion sensors, especially at lower motion amplitudes (Ref. 7).

A large number of engineering simulation laboratories have motion systems as part of their equipment. Some of the most important or most powerful helicopter simulation facilities and their motion system performances are (Ref. 13):

NASA Ames Vertical Motion Simulator (VMS)

The NASA Ames Research Center Vertical Motion Simulator is shown in Figure 7 (Ref. 10). The present cab is of the same specifications as the FSAA (referenced below), but is driven in angular motion by a small, six-actuator hydraulic system. This is mounted on a laterally driven carriage with 13 m of travel atop a beam which can be moved vertically in a 19 m envelope. These latter two drives are electric, and are capable of nearly 1-g accelerations. A second horizontal motion component is not provided; however, the cab can be rotated to substitute fore-and-aft motion for lateral motion (Ref. 11).

NASA Flight Simulator for Advanced Aircraft (FSAA)

The Flight Simulator for Advanced Aircraft features a lateral motion envelope of 30 m, together with 3 m of vertical travel and 2.5 m of fore-and-aft movement. Three independent drives provide generous amplitudes of angular motion. All drives are electric. Linear acceleration capabilities are modest, less than ± 0.5 g, but are generally satisfactory for helicopter simulation. The large transport-type cockpit has two pilot stations, and is equipped with hydraulic control loaders, visual simulation TV monitors, and head-up display equipment. This cab is reconfigured for each new simulation. Over the past decade, this facility has been used in simulation of a wide range of aircraft. Currently, helicopter simulations make up about 25 % of its operation (Ref. 11).

RAE, Bedford, U.K. Advanced Flight Simulator (AFS)

During 1985, the present RAE simulator (Figure 8), which comprises a Sigma-8 digital computer, an AD-4 analogue computer (used for interfacing and driving analogue components such as instruments, chart recorders, motion system, and TV visual system) and single and two-seat cockpits mounted on their respective motion systems, will be retired from use. They will be replaced by the Advanced Flight Simulator, which will use a network of SEL 32/87 and 32/27 computers, a digital data highway, a large amplitude 5-axis motion system with interchangeable cockpits and improved visual systems. The AFS motion system will provide improved motion cues in five axes, including up to 1-g transient heave acceleration. The interchangeable cockpits will include a dedicated helicopter cab, and vibration and sound cues will be significantly improved (Ref. 12).

Visual display systems. Major requirements for visual simulation of out-of-the-window real-world scenes derive from two different factors: Firstly, the overwhelming performance capabilities of the human eye, and secondly the operational flight regimes and missions of the respective aircraft that is to be simulated.

The eyes are the most important sense organs for gaining information about the world around us. However, it is largely unknown how the visual information is processed and manipulated by the human brain. It is estimated that over 90 % of the information that we receive during our normal daily activities comes through the eyes and certainly that much, or more, when an aviator is involved in flight tasks (Ref. 14). The fantastic characteristics of the human eyes are represented by the visual resolution capability of about 1 arcmin, combined with a field-of-view that is practically unlimited due to the free mobility of the pilot's head and body (provided no other factors such as narrow cockpit dimensions or canopy frame visual interferences are more restricting). This performance is practically impossible to meet in simulation, even by the most sophisticated visual display systems being planned, so, as with the motion cueing, visual simulation entails a compromise with reality.

The required characteristics of visual simulation systems have been addressed in numerous publications (see Ref. 14), differentiating between :

- spatial properties (field-of-view (FOV), scene content, optical range)
- energy properties (luminance, contrast, resolution, colour)
- temporal properties (picture refresh rate, time lags).

Most of these factors are interrelated and interdependent, as is oftentimes true in engineering processes, so we shall pick out just the most illustrating ones here.

Simulator visual system architecture may be described as consisting of three elements in series: (1) an information storage subsystem; (2) an information retrieval and processing subsystem; and (3) an information display subsystem (Ref. 15). The two most commonly used information storage systems are based either on terrain model boards or on computer generated imagery (CGI).

Model board systems have successfully been in use in R&D simulators as well as in training simulators for quite a long time. They are composed of a three-dimensional physical model of the terrain over which the simulated aircraft missions are to be executed. Translational motion, corresponding to the motion of the simulated vehicle, is provided by a servo-driven colour television camera which is moved over the model board, and, generally, rotational freedom is realized by rotation of an optical probe. Scene display to the simulation cockpit is provided by television monitors or projectors.

Model boards like the examples given in Figure 9 or the unique V-notch hurdles in Figure 10 (Ref. 16, scale 700:1) offer the richest scene content and texture detail but apparently have physical limitations on operating volume. Ref. 10 states that probably the smallest scale that will allow a sufficiently high quality picture for NOE operations is around 500:1, and a rough calculation may be used to prove this: Given an absolute vertical height clearance of the optical probe system of about 0.5 cm above the model board, the minimum simulated pilot's eye height will be 2.5 m. This is too great a value for a number of existing helicopters (e.g. pilot's eye height in a MBB-BO 105 is 1.7 m). Another limitation in the use of model boards may be the requirement for deteriorating visibility, which can only be achieved by means of subsequent picture processing.

NASA Ames operates two TV model-board visual scene generators. These systems can provide a 34^{o} by 48^{o} visual field on a 525-line colour television in raster format. The model boards have accumulated a variety of features modelled at scales from 300:1 to 1200:1 (Figure 11, Ref. 11). The Boeing Vertol flight simulation facility (Figure 12, Ref. 17) is equipped with a four-camera wide-angle television/terrain model visual display system for the terrain flight under visual meteorological conditions, having a four-window cockpit visual display covering a FOV of $125^{o} \times 75^{o}$. Various other institutions have adopted the techniques of terrain model boards, but it would go far beyond the scope of this lecture to name all of them.

The simulation of visual informations has made tremendous progress by the introduction of computer generated imagery. With the recent development in high-capacity mass-storage systems and high-speed operating systems, the primary restrictions for sophisticated visual simulation now originate more from the display systems than from limited computer capabilities. The problem lies in the natural law that a horizontal or vertical extension of the FOV inevitably leads to a reduction in display resolution, provided the display generators and projection systems remain unchanged.

High-resolution wide-angle fields-of-view are required in particular when we consider the helicopter-specific flight regime. Helicopters typically fly low and slow, and "Nap-of-the-Earth" (NOE) has become a well-known term to describe tactical point-to-point flying and hover operations in close ground proximity (Figure 24). Moreover, steep take-off and landing procedures also constitute typical helicopter flight profiles. The environment for the pilots flying these missions is rich in detail, and terrain features, as well as the visibility factors of weather and darkness, are elements of the environment that may significantly affect the helicopter pilot's task (Ref. 3).

Other vital demands for a large FOV are posed by the outstanding ability of helicopters to yaw rapidly. Based on pilot's minimum preview times for obstacle avoidance, and dependent on the actual yaw rate as well as on the roll attitude, the visual scene must cover obstacles that are some degrees ahead in the projected flight path azimuth. And, last but not least, piloted simulation of quick stop maneuvers for a fast transition from forward to hover flight or the recovery from autorotation is almost unacceptable when visual cues get completely lost at large nose-up pitch attitudes because of the limitations in the vertical FOV.

In the MBB fixed-base simulation facility in Munich-Ottobrunn, FRG, a 3-channel CGI visual system has been installed, presently offering a FOV of 26° vertically by 106° horizontally, using a beam splitter projection system (Ref. 4). The cockpit now available for helicopter simulation is a BO 105 prototype fuselage. The dual-seat cockpit is fitted with the original instrument panel, including original and modified flight instrumentation. Figure 13 shows internal cockpit details and the external view of the CGI-scene. The out-of-the-window environment is composed of contour images in correct perspective, such as the scene in Figure 14, which shows an existing airfield area in Germany, including the runway, buildings, and the surroundings ranging 50x50 NM. Other interesting features simulated are an area for executing slalom tasks, built as a course of red-and-white poles (Figure 15), and a dolphin flight test course with a number of plate obstacles forming a row (Figure 16).

The basis of the image generating system is a General Electric CGI-COMPU-SCENE II. The system allows the generation of up to 8.000 edges maximum per scene. Various visibility conditions like haze, fog, clouds, and day- and night-time may be simulated. Additionally, moving objects (targets) may be inserted.

In late 1986, a General Electric 5-channel COMPU-SCENE IV dome projection system will be available (Figure 17), with optional expansion capability to 9 channels. Compared to the former visual system, a considerable extension of the overall FOV will be achieved. Figure 18 shows the FOV for a tandem cockpit as it is designed for the new generation MBB anti-tank helicopter PAH2. Apart from the fact that the FOV is extended to 115° vertically by 140° horizontally (extension capability 150° x 300°), highly detailed terrain features and texturing can be displayed (Figure 19). Here a photomapping method is used, filling edge-defined contour areas with cell textures gained from photographs of significant features, like terrain patterns, or trees with distincly visible leaves. Textural details of moving objects, such as trucks, tanks, smoke, may also be portrayed. Through the use of these separately processed textures, the number of edges and the time needed to compose detailed scene features is reduced substantially. The simulation fidelity obtained of low level flight cues is most amazing, and, after all, in the light of the new techniques, the older generation visual scenes (Figure 20) now appear rather like animated cartoons.

Both Hughes Helicopters and Sikorsky Aircraft will use COMPU-SCENE IV in their simulation facilities (Ref. 18). NASA Ames will complete their Rotorcraft Systems Integration Simulator (RSIS) with a new interchangeable rotorcraft cab on a four-degrees-of-freedom Rotorcraft System Motion Generator (RSMG), and an advanced visual system in the Advanced Cab and Visual System (ACAVS) (Figure 21, Ref. 10). The total system is later to be moved as a whole onto NASA's Vertical Motion Simulator (VMS). The RSIS dome projection system will supply at minimum a FOV of 120° by 60° and is planned to cover 240° by 180° later. The system is scheduled to be fully operational by the end of Fiscal 1986 (Ref. 19).

For all existing and future visual systems, the compatibility with pilot night-vision aids, such as night vision goggles (NVG), and non-visual sensor equipment, such as FLIR (forward looking infra-red), is to be demonstrated. Figure 22 shows the current CGI-scene in the MBB-simulator, operated with a helmet mounted NVG at twilight (Ref. 37). Although the visual contrasts were slightly reduced relative to the normal view, operation with NVG in a CGI-simulator was satisfactorily possible.

Following the technical assessment of the visual simulation systems, given in Ref. 10, the newest and most advanced visual concept, feasible for future rotorcraft simulators, could be the helmet mounted display (HMD) concept (Figure 23). In this concept, a small virtual imaging system is mounted on a crew member's helmet. The images from three light-valve projectors are processed optically into two scenes (one for each eye) at the output of the projectors. A head tracking system provides information on the pilot's head position to the CGI-system. There are numerous advantages of this HMD concept, such as effectively unlimited field-of-view, minimum distortion, and a large space and weight saving (Ref. 10). The artist's concept as shown in Figure 23 may give an indication that computer generated imagery still provides potential for rapid advances in future visual systems technology, and, as a matter of fact, we can assume that this technology will help to completely eliminate the technical limitations of current ground-based simulators.

2.2 USE OF GROUND-BASED SIMULATORS

Generic studies of handling-qualities. In recent years, helicopter missions have placed considerable emphasis on terrain-flying tactics (NOE) for survival and high combat effectiveness. (Figure 24). These tasks place strong demands on the performance, agility and precise control characteristics of rotorcraft. It is widely known that current helicopter flying qualities specifications (MIL-H-8501A, MIL-F-83300) are based on an absolute standard, and do not address specifically today's requirements on the performance of the pilot-vehicle combination in mission-oriented tasks. In many instances, they can result in undesirable flying qualities. Therefore, a fundamental understanding must be established of how the handling qualities influence the performance of mission tasks and a data base must be provided for development of new handling qualities criteria for optimum flight path-management, allowing the design of better aircraft.

Fundamental studies of generic handling qualities effects by ground-based piloted simulations were conducted by Chen, Talbot (Ref. 20). Their investigations explored the effects on terrain flying qualities of rotor design parameters (flap-hinge-offset, Lock-number, pitch-flap coupling), interaxis couplings and various levels of control and stability augmentation. The parametric studies covered the full range of teetering, articulated and hingeless rotor system families. The simulation experiments were conducted both on fixed-based and moving base simulators, and were later extended also to in-flight simulations. The scope of tasks included longitudinal/vertical (dolphin) tasks, lateral slalom tasks, and combined tasks. As a result, Figure 25 shows a summary of piloting ratings for various combinations of damping and control sensitivity in the roll axis, as obtained from the various simulation experiments. It is noted that low control sensitivity and angular rate damping combinations were found to be unacceptable, and minimum levels of damping ($L_p > -5$ sec^{-1}) and of control power ($L_{\delta_a} > 1.5$ $^1/s^2$/in) seem to be necessary for pilot acceptance. Moreover it was found that other factors may degrade the control effectiveness inside the satisfactory boundaries.

Another area of particular importance for helicopter handling-qualities are the effects of pitch-to-roll and collective interaxis couplings. The ratio of the roll moment resulting from pitch rate to the roll moment resulting from roll rate (L_q/L_p) plays a dominant role in the short-term aircraft control response. Figure 26 shows a clear and substantial variation of the pilot rating with L_q/L_p, as obtained by Chen's piloted simulation experiments (Ref. 21). The boundaries indicate an unacceptable flight behaviour, if the coupling value exceeds a value of 0.35; values greater than 0.5 seem to imply unacceptable flying qualities. For the nap-of-the earth operations it is equally important to understand more in detail and to quantify the influences of the vertical axis characteristics of helicopters on the overall handling qualities. These effects, largely undefined until recently, were adressed during large scale motion simulator experiments on the Vertical Motion Simulator (VMS) at Ames (Ref. 22). The effects of vertical axis characteristics (damping and collective control sensitivity), of engine governor dynamics, of the rotor inertial energy, and of the engines excess power were particularly investigated. Figure 27 indicates the variation of pilot rating with vertical damping and collective control sensitivity. It is seen, that a certain level of minimum damping, and a considerable amount of collective control sensitivity ($Z_{\delta_C} = 0.4$ g/in) is needed to achieve acceptable piloting rating. It was also found that variations in the engine governor response time can have a significant effect on handling characteristics. For a slow governor, the degradation in pilot-rating in NOE-tasks can be as much as two ratings.

Chen and Lebacqz (Ref. 23) also report from piloted simulation experiments investigating the effects of variations in longitudinal static stability. Static stability with respect to angle of attack (M_w) and speed (M_u), and the resultant variation of longitudinal stick gradient versus speed (at all other controls held constant) are of particular importance in view of airworthiness/acceptance (Ref. 24). Figure 28 shows Cooper-Harper ratings from various simulation experiments, as a function of longitudinal stick gradient (for unstable cases the inverse of the time-to-double amplitude of the unstable root is drawn). The data show a slight trend toward a degraded capability as the stick stability is reduced. In terms of Cooper-Harper rating, however, the aircraft have still been rated as adequate for the given task, irrespective of the level of static stability.

It can be concluded from the examples shown above, that piloted ground-based simulation is a highly valuable tool to evaluate the effects of configurational design parameters and to provide a reliable and useful database for the design of better aircraft configurations. The lack of mission-oriented handling qualities data in the current specifications can be compensated.

New control systems development. New tactical requirements for battlefield operations are likely to place an increasing emphasis on performance and agility during NOE-tasks. Operation in poor visibility or darkness, made technically feasible by advances in sensor technology, will further increase the demands on the pilot. The development of battlefield-compatible advanced flight control systems will decrease pilot workload and improve the aircraft handling qualities, and will thus allow the pilot to remain able to exploit the full potential of the next generation of helicopters.

As was stated before, current handling qualities have failed to provide more than basic guidelines to industry. Criteria for designing advanced helicopter flight control systems for military missions do actually not exist. Therefore, subjective ratings during initial design studies must be obtained in a man-in-the-loop ground-based flight simulator. Consequently, simulation in the future will be a central part within the design stage of cockpit controllers and advanced flight control systems. Thereby the simulator can be used to evaluate both the individual elements of control and display systems, and the interactive effects of such systems.

Extensive simulation work was conducted in support of the U.S.-Army's Advanced Digital/Optical Control System (ADOCS) program at the Boeing-Vertol simulation facility (Ref. 17). Piloted simulations served to assess the interactive effects of : (1) Sidestick-Controller configurations with different levels of integration, (2) Automatic Control and Stability Augmentation Systems (CSAS), and (3) different types of Displays and Vision Aids, both during day VMC and night IMC-conditions. During the evolution of such elements, it is important to investigate and find those combinations that produce the optimum level of handling qualities ratings, rather than to optimize each component for itself.

Figure 29 shows various types of side-stick pilot controller configurations investigated including different levels of integration (number of axes to be controlled) and showing different force-/deflection characteristics. As a result, clear choices were made for small-deflection separate sidestick controller configurations ((3+1) collective), definition of various generic SCAS configurations were made, and principal degradations in pilot ratings were worked out, when maneuvering tasks were conducted unter IMC conditions.

Careful design and tailoring of both the basic aircraft mechanical control system and, moreover, of the automatic flight control and stability augmentation system is of major importance for future aircraft, since the extremely difficult task of flying NOE at night will leave the pilot little capacity to his role of managing the flight path. An ideal solution would be to completely decouple the stick response characteristics where 'nuisance' axis couplings are evident, and to some extent this is achieved by the conventional SCAS. However, some error will always occur; since a deviation is required before for the feedback loop is able to function. Secondly, safety considerations normally dictate a limited, typically ± 10 %, SCAS authority so that complete axis response decoupling is not practicable using the SCAS alone. Mechanical cross coupling of the controls provides a solution, but owing to the constructional complexity is limited to a simple constant or function with stick position. A complementary design uses a fixed mechanical coupling for the basic solution with the SCAS providing the fine tuning. To investigate this potential simulation experiments on the optimization of the primary mechanical control system couplings were conducted at the MBB simulation facility; the evaluation was accomplished using special terrain avoidance obstacles in the CGI-scene (Figure 16), and testing various low-speed, high-speed and nap-of-the-earth tasks. The object of these tests was to investigate the control coupling design methodology and to establish pilot opinions. Figure 30 shows an example of the strong reduction in the unwanted yaw response in a dolphin maneuvre achieved through cross feeding collective inputs to the tail rotor collective at the rate of 1 and 2 degrees/degree. Clearly the yawing motion is reduced, as the coupling is increased. A primary goal of the pilot in-the-loop was to confirm the design methodology, which is based on computing the ideal response from the stability and control derivates as shown in Figure 30. The pilot rating was substantially improved by introduction of the collective/yaw control cross coupling, however, over-compensation as shown in Figure 31 was undesirable as the helicopter's reactions then became unnatural. The 1 degree/degree was established as a compromise solution for the basic mechanical coupling with the over or under compensating difference built into the SCAS.

Collective to longitudinal cyclic control coupling presents a more complex design situation as the basic helicopter response characteristics are strongly dependent on the forward speed. The simulation work once again substantiated the design methodology as shown in Figure 32 , but pilot's opinion rapidly deteriorated as the control coupling deviated from the design optimum. Particularly, irritating for the pilots was the overcompensation in pitch response to collective if the gain was higher than that required. In an attempt to maintain a satisfactory but simple hardware solution a concept was developed the mechanical couplings were optimized for the typical NOE speeds of around 60 kts with the AFCS compensating for the response characteristics which were speed dependent. Most revealing were the investigations of the collective to roll coupling. Unlike the collective to yaw and collective to pitch. The initial direction of roll response is influenced by the rotor flapping frequency ratio and, more importantly, the blade Lock's number. The rotor system under investigation on the simulator causes an initial roll left in response to collective up inputs, though the final response is always roll right. This initial roll response, however, was not at all disturbing to the pilot, and the simulator results in fact demonstrated that a collective roll control coupling to compensate for the secondary and dominating roll right response was more desirable from a pilots point of view, since the secondary response is also influenced by the pitching motion of the aircraft. The optimum collective-lateral cyclic coupling will also depend on the control couplings established for the other axes.

One further important objective and scope of simulator work is to investigate, by means of ground based piloted simulation, and to optimize control laws for automatic control and stability augmentation systems, trim/force augmentation systems or autopilots. The effects of various types and levels of stability and control augmentation on the flying qualities and agility of helicopters in terrain-flying tasks were investigated during various piloted simulation experiments (Ref. 16, 17, 21). Fundamental investigations showed that both rate-command and attitude command CSAS made substantial improvements in terrain-flying qualitives in otherwise unacceptable helicopter configurations. The feedback of the pitch rate and pitch attitude to collective pitch and to the longitudinal cyclic pitch was found to be favourable especially for hingeless- and stiff-hinged-rotor helicopters. It was also found by such experiments (Ref. 17) that strong interactive effects of sidestick-controller characteristics and stability and control augmentation on handling qualities judgement exist. In other words: High level of control augmentation may require controller characteristics that are different from those, if a much more reduced SCAS-level is provided. Continuation of piloted simulation studies on this subject is highly required, with main emphasis given to the requirements of control laws, improvement of pilot rating in IMC-conditions, and investigations of the effects of turbulence on system performance and pilot workload.

Active Control Technology, having been well demonstrated for fixed-wing aircraft, is now likely to be accelerated also for helicopters. However, the application to these vehicles is presenting new problems, since helicopters are dynamically more complex. Fundamental research work in this area is being carried out by the UK Industry and the Royal Aircraft Establishment, RAE (Ref. 16, 25). Piloted, ground-based simulation is used to first assess the relative merits of a range of different control schemes in a so-called "conceptual" helicopter model . This conceptual approach allows a comparatively quick and simple means of evaluating likely relevant control modes and of identifying the optimum control strategy. These results are typified in time histories, as shown in Figure 33. With appropriate criteria established, the final control law design is then realized and its benefits are confirmed on the most representative aircraft model available. These simulation experiments are conducted using a number of different assessment features, like the serpent course (Figure 9) and the unique "V" notch hurdles, as shown in Figure 10. Undoubtedly, with the progress in small digital computers, active control technology on helicopters including more and more complex control laws will give a major advance in future helicopter operations. This development will rely strongly on ground-based piloted simulation.

To support the development efforts, the ground-based simulator can also be used to investigate and reconstruct accident situations and to simulate critical system failures. The helicopter behaviour following a hardover signal of automatic flight control is one important question which must be answered in an early stage of such system developments.

Engine failure during critical flight phases (Figure 34) is another topic which has to be treated in any helicopter developmen. program. With the engine and engine governor dynamics incorporated into the mathematical model, the ground-based simulation can be used to investigate aircraft behaviour, pilot reaction and emergency procedures after a partial or total loss of engine power. Demonstration of the height-velocity (H-V) diagram assuring flight regimes from which a safe landing after engine failure may be executed is of particular importance during certification procedures. One-engine failure during take-off and landing of CAT A transport category aircraft is a vital event, and the remaining take-off and landing distances in such a case is one main issue in the discussion of the flight evenlope for such aircraft. With the improvements, especially in visual presentation, the flight simulator can be used to investigate the system handling and flight path management, and to enable the pilot to cope with these situations. And there is also no doubt that ground-based simulation will play an increasingly important role during certification procedures, and will step-by-step be accepted as a valuable means of demonstrating compliance with existing certification regulations.

Simulator use in display and control sophistication. Spectacular technology advances during the next decade will inevitably be dominated by the enormous progress in electronics, micro-processor computing, and the associated avionics and optronics. One area of future helicopter design, where electronics and optronics will have a major impact, is in the cockpit. Figure 35 shows a cockpit arrangement of a current generation aircraft featuring a diversity of controls, displays and instruments, resulting in substantial pilot workload, extensive and costly training, and reduced mission effectiveness.

To eliminate these shortcomings, substantial efforts are currently undertaken to provide and integrate future airborne avionic and electronic equipment into the cockpit. Some of the most important systems technologies are subsequently mentioned. Multifunction display technology, doubtless, represents the most outstanding innovation and opens the most promising degree of freedom in cockpit design. Thereby one of the key elements to improve cockpit layout and organization is the integrated control and display unit (CDU). It combines a number of functions for which normally a variety of different control units have to be used. In addition, fully integrated control capabilities transfer the majority of control and switch functions away from the panel into combined cursor and touch controls, allowing a significant workload reduction for the crew. Recent research in speech synthesis even offers promise for voice-command and display systems in future application,at cost for caution and warning displays. This means, that current avionic integration methodology will result in highly integrated, coordinated aircraft systems (Figure 36).

As an example of current cockpit integration/simulation work done at MBB, Figure 37 shows a complete cockpit-mockup of a light civil helicopter, including modern control and display technology in a highly integrated cockpit arrangement. The functional verification will take place in the CGI-Simulator. The cockpit mockup takes the form of a fully representative cockpit interior, and will be used for control and display units assessment, subsystem controls methodology studies and for aircrew mission tasks, in general. The mockup will be linked to the Flight-Dynamics and CGI-Simulator (Figure 17) so that the crew can fly the fully representative Helicopter Model under realistic piloting issues, in a realistic scenario.

The prime use of simulation as a design tool at this stage is to demonstrate the feasibility of unproven cockpit technology and to test these concepts for acceptability and operability. Ground-based avionics simulation provides a tool which allows designers to analyse the implications of their design options in the proper context, and understand the inteactions between the pilots and the individual systems. It allows practical testing of hardware in-the-loop as well as verification of related software. It is the main benefit of ground-based avionics simulation that this can be done without the complexities and safety implications of airborne testing, in a real mission environment, well before first aircraft flies.

Man-machine integration. In the last decade especially the military operator has considerably expanded its use of the helicopter. The potential is appreciated to conduct not only the hitherto traditional missions, but also to take on new types of missions, like land combat functions, intelligence, firepower, combat support, and command communication (Ref. 10). Simultaneously, the helicopter has also acquired new tactics and new performance requirements resulting in a tremendous increase in the number of subsystems. The aircrew is expected to maintain a high degree of control and mangement over all these systems, while flying close to the ground, and possibly at night and in poor weather conditions.In this respect, modern aircraft and systems technology will help to make the helicopter more easily to fly, however, the increased complexity and sophistication could lead us to the limits of the human pilot's capability.

Simulation provides an ideal tool to realistically examine the interface between the crew and the avionic systems under a realistic representation of the cockpit function, and in a representative environment. As an example, some of the investigations required to assess system handling throughout complete missions include: communication, navigation and other basic procedures, target acquisition and weapon systems operations, interaction with computer terminals, flight management and diagnostics control, and crew work-sharing. Furthermore, since future helicopter missions are also to be flown at night and in adverse weather conditions, it is necessary to consider also the impact on handling qualities of the pilot's night vision aids (Figure 38), such as night vision goggles, HUD and IHADDS. These systems have to be integrated and made compatible with flight instruments and displays.

In addition to the outside world visual scene, it will be necessary to display to the pilot an easily understood image of the tactical situation and navigation functions. The achievement of this capability will require the development of real-time tactical scenario displays, which can be used to investigate the man-machine interference required for battle-captain functions such as target engagement and threat defense. Such programs will be an interdisciplinary effort involving pilots, display engineers, mathematicians, and engineering psychologists (Ref. 26).

Closed loop integration testing. As weapon systems complexity increases, more and more rigorous capabilities of sub-system and total system integration are necessary. Once the modern simulation facility is established, the simulator can successfully be extended to assist in system integration work on the integration rig (Figure 39). The original equipment, integration computers and software can be linked via available interfaces, thus allowing closed-loop integration testing. The integration rig interface usually requires fast handshaking between the simulation computer and the integration computer to allow fast transfer of dynamic data during real-time closed-loop investigations. The hardware-in-the-loop helicopter airframe is connected to the computer facility in order to help to evaluate the individual systems. The actual cockpit is used in the CGI-simulator for inclusion in piloted real-time simulation.

It is the interactive effect of all those elements and disciplines that requires a new generation of advanced ground-based simulation laboratories and facilities. Some existing simulation capabilities, such as at Ames (Ref. 10), are almost exceptional, and will be further augmented when improved simulator components become available. Due to the increased requirements to support the designs of next-generation rotorcraft development programs (PAH-2, LHX, IVX), also the helicopter industry is spending large-scale investments in improved simulation facilities with advanced visual presentation, to support engineering, man-machine interface and integration efforts.

3. IN-FLIGHT SIMULATION

In the foregoing sections the principal elements of the ground-based piloted flight simulator have briefly been reviewed and the most important application in view of future developments in aircraft and systems design and criteria development have been discussed. As a matter of fact, simulator technology is getting more and more specialized, and the "synergistic" capabilities of sophisticated motion systems and wide-angle computer generated imagery (CGI) visual systems have made the ground-based simulator a primary and highly effective tool. However, in spite of the improved sophistication of these simulation facilities, there are numerous specialized applications where unrestricted motion, three-dimensional visual information and the "true"operational environment are crucial to the success of the simulation. In those cases, where the ground-based flight simulator still has inherent limitations, the airborne flight-simulator offers the only alternative.

It is the purpose of this part of the lecture to review briefly some of the considerations involved in in-flight simulation technology and to discuss the unique role of airborne simulation in the different phases of research, development and operations.

3.1 IN-FLIGHT SIMULATOR HELICOPTERS

Extensive experience with the development and operation of in-flight simulators based on helicopters as well as fixed-wing aircraft have particularly been made in Canada, in the United States, and also in Germany. To give an overview, the variable-stability helicopters which have been used in the past and are likely to be used for future applications are briefly characterized. The aircraft discussed are the CH-47B helicopter, the UH-1H V/STOLAND helicopter, the RSRA aircraft, the BO 105 S3 in-flight simulator, and the BK 117 HESTOR (planned project).

The NASA/Army CH-47B is a twin-engine tandem-rotor cargo helicopter, capable of lifting a 10,000 pound payload. The basic aircraft system includes hydraulic boost actuators and dual series actuators for the Stability Augmentation System (SAS). The large speed range (up to 160 kts) of this aircraft is particularly attractive, and the fairly high control authorities in pitch and roll implies the capability of simulating the trim characteristics of a wide range of helicopters. More aircraft details are given in Ref. 27.

The NASA/Army UH-1H (V/STOLAND) helicopter (Figure 40) is a single-engine, teetering rotor aircraft used for variable flight control and display research. The aircraft had been modified by adding an avionics system, called V/STOLAND. It has parallel electromechanical actuators in 4-axes with nearly 100 % authority, and series electro-hydraulic actuators with limited (20 % to 30 %) authority. Flight research tasks included investigations of flying qualities for low altitude agility maneuvering. However, the application as in-flight simulator is limited by the inherently low control power of its teetering rotor system. More detailed descriptions are given in Ref. 27.

The canadian NAE Bell 205A-1 is the civil equivalent of the UH-1H. The aircraft is equipped with full authority dual-mode hydraulic actuators, which provide full-authority electrical fly-by-wire control from the simulation pilot's seat. The rotor stabilizer bar is removed, to improve the rotor cyclic input response. The 205A-1 simulator aircraft, at present, is the superior vehicle for general flying qualities research. A detailed description of the Bell 205A-1 airborne simulator aircraft is given in Ref. 28.

The NASA/Army RSRA (Rotor System Research Aircraft), Figure 40 , is intended for comprehensive in-flight testing and verification of new rotor system technology (Ref. 27). The aircraft is designed to fly as pure helicopter, compound helicopter, or as fixed-wing aircraft. The many uses of this aircraft are aimed to principal areas of rotor research: Evaluation of rotor characteristics over a wide range of operating conditions, determination of aerodynamic characteristics (rotor derivatives), measurement of rotor noise characteristics, to name just a few. The aircraft is equipped with an advanced fly-by-wire digital control system to provide various rotor control systems and adequate handling qualities with rotors of uncertain characteristics. This system has full authority control of the rotor and fixed-wing surfaces.

The BO 105-S3 variable stability helicopter (Figure 40) was developed and operated at MBB since 1975 as a flight simulator for flight control and guidance systems (Ref. 29). In-flight simulation programs included evaluation of advanced digital control systems, remote helicopter control, and development and test of advanced controller and flight guidance systems. The aircraft is now operated at the german DFVLR Braunschweig and has been equipped to the in-flight simulator BO 105 ATTHeS (Advanced Technology Testing Helicopter System), Ref. 30 . The BO 105 is a twin-engined, light-utility-class helicopter, equipped with the fourbladed "hingeless" rotor system. The simplex fly-by-wire control system includes full-authority, quick response hydraulic actuators in 4-axes, mechanically connected to the safety

pilot's controls. The inherently high control power and damping characteristics about the pitch and roll axes, provided by the hingeless rotor, allow simulation of a uniquely broad range of helicopter characteristics.

The realization of an operational in-flight simulator "HESTOR" (Helicopter Simulator for Technology, Operation and Research)on the basis of the BK 117 aircraft is planned in cooperation between DFVLR Braunschweig und MBB. The application of this aircraft is mainly intended for tasks in the areas of development, testing and integration of new technologies in advanced control and stabilization systems, new cockpit controller and display technologies and active control technologies for vibration reduction, and general flight characteristics imprcvement (Ref. 31). The BK 117 aircraft is driven by the hingeless rotor, and has high performance engine power installation. Thus, the modern aircraft design provides sufficient performance capability for an extended flight envelope, and suitable cockpit environment compatible with mission requirements for 1- and 2-man cockpit evaluations.

Both in-flight simulator aircraft, the BO 105-S3 and the BK 117-HESTOR provide adequate compatibility with the ground-based MBB-SGI-simulator (Ref. 4) so that in-flight validation of ground-based results can easily be made.

3.2 REQUIREMENTS FOR IN-FLIGHT SIMULATORS

The scope of simulation of the flight regimes and flight behaviour of other aircraft is defined by the own characteristics of the basic simulator aircraft. There are several principal capabilities which need to be provided by an in-flight simulator to fulfill the needs of other aircraft design (Ref. 32). Some main aspects, which are of particular importance under aeromechanical aspects, will be briefly discussed in the following section.

Basic helicopter characteristics. Some of the most important requirements for in-flight simulators in providing the desired envelope result from sufficient control power and flight performance capabilities. In order to adequately simulate the static and dynamic response of a broad range of other helicopters it is required that a sufficient range of control power, control sensitivities and response characteristics is available (Figure 41). This includes control power in each axis and low time lags in the control reaction. High control power is needed when simulating helicopters with low order augmentation system. With such systems, a wide variation in the dynamic characteristics of various helicopter concept exists, as high control power is needed to simulate this range of characteristics. In this respect, the inherently high control power of hingeless-rotor helicopters is especially suitable for in-flight simulator operation (Figure 41). Short control time constants in both axes also allows studying of high gain following control and stability augmentation modes. Furthermore, short control lags alleviate design of control actuators with proper dynamic characteristics.

Mathematical model representation. An essential element of a flight simulator of any kind is the mathematical model representing the vehicle to be investigated. Past experience has shown that successful use of in-flight simulation requires the capability of isolating unwanted aircraft responses and to divorce the simulator's inherent aerodynamic characteristics. For investigations of more generic flying qualities problems the importance of accurately representing specific dynamic response characteristics may not be of great concern. However, if a specific design must be simulated, or higher frequency modes of motion or turbulence response has to be represented, a high level or variable-stability control system is required.

The requirement for isolating certain aircraft responses or changing basic aircraft characteristics requires the application of "model-following" techniques. A fundamental comparison of response-feedback and control-feedforward techniques, and of model-following systems has shown that there are major advantages in using model-following techniques (Ref. 32). Such techniques allows to implement a standard equations-of-motion model, and to quickly and easily incorporate the aerodynamics of the simulated vehicle for each program. When selecting the modelling technique, computational capabilities must be taken into account, since imperfections in the model can strongly influence the validity of the in-flight simulation.

For realization of variable stability capabilities of the BO 105-S3 in-flight simulator, a model-following control system (MFCS) was designed and tested by the U.S. Army and the DFVLR (Germany). The block diagram of the MFCS is presented in Figure 42 . In a typical model-following control system the pilot's commands are disconnected from the actual aircraft and fed into a model. The errors between the states of the model and those of the base system are fed into a controller matrix, which attempts to minimize the state errors by generating control signals for the actuators. The controllers used are feed-forward gains, its elements are adjusted with airspeed to improve accuracy and robustness of the system. A detailed description of the MFCS design and ground-based simulation testing is given in Ref. 33.

To demonstrate the performance of the MFCS, simulator experiments were first conducted on a NASA ground-based simulator, showing significant improvements in task performance and handling qualities for different helicopters with both teetering and hingeless rotor systems. First flight test results (Figure 42) demonstrate the effectiveness of the control technique. The measured time histories illustrate a decrease in interaxis coupling, resulting in a substantial reduction of pilot control activity. With the augmentated BO 105 a rate of climb flight condition can be initiated only by collective pitch control input (Ref. 30).

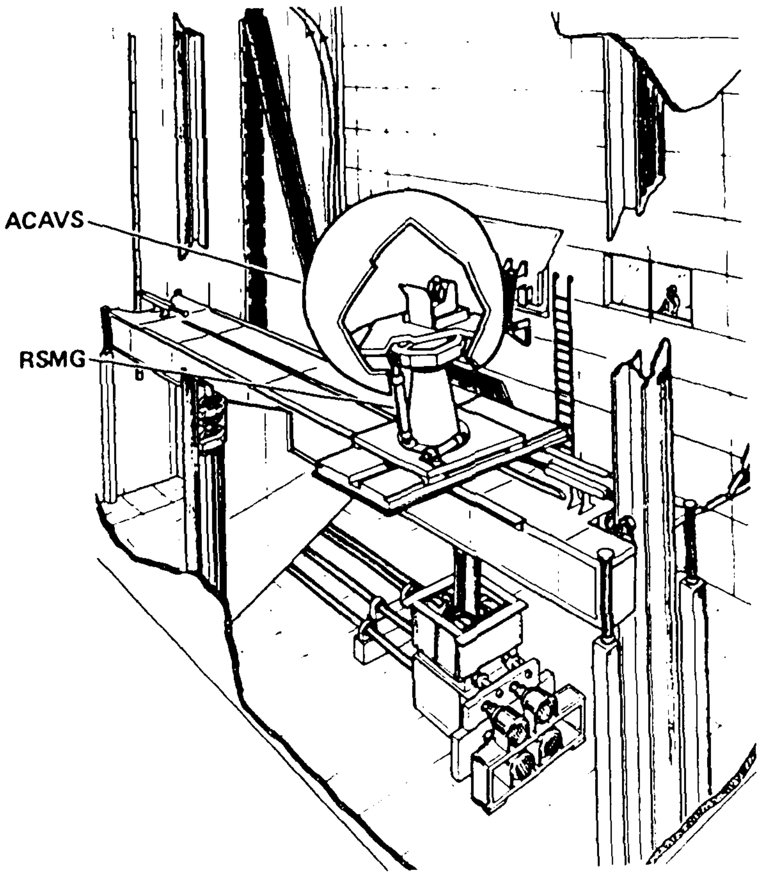


Figure 21 Future NASA Ames VMS/RSIS project overview [10]

Figure 22 CGI with a helmet-mounted NVG at twilight [37]

Figure 19 Typical Compu-Scene IV environment

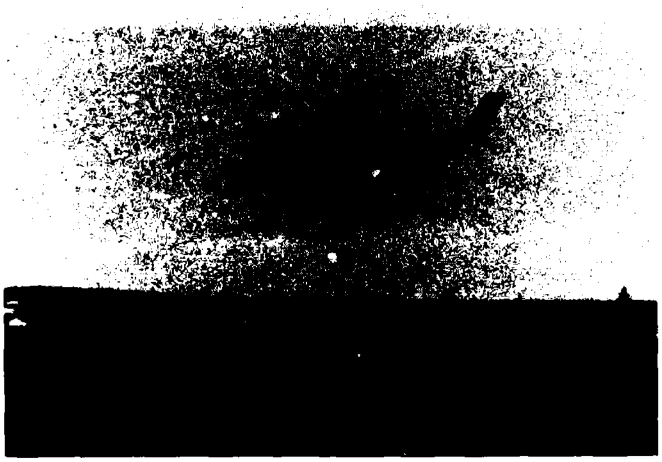


Figure 20 Compu-Scene II visual scene

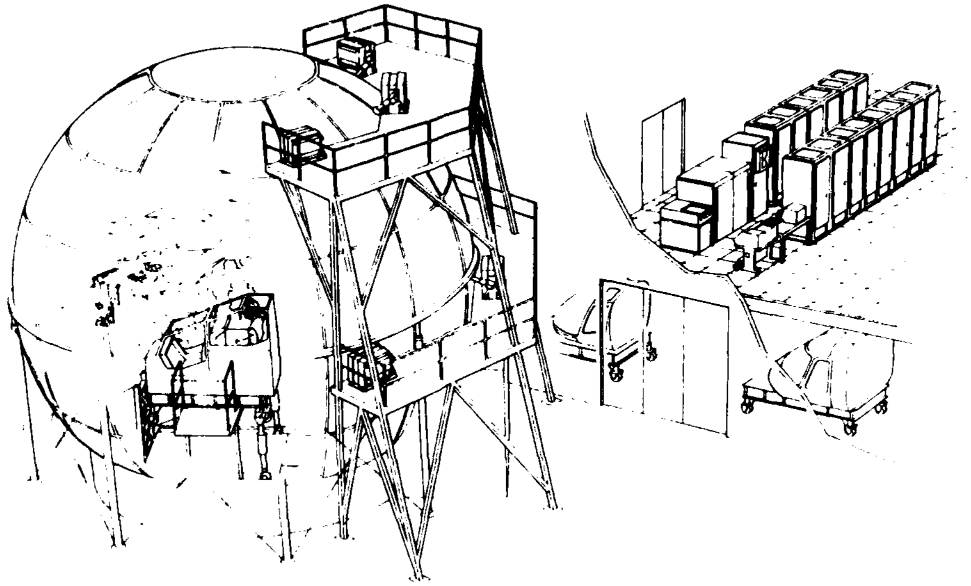


Figure 17 Extended MBB ground-based simulation facility

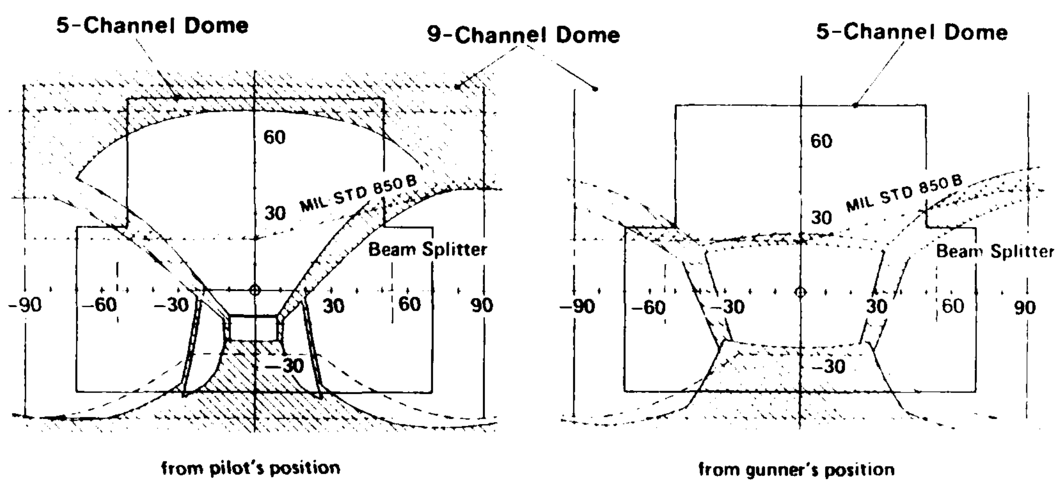


Figure 18 Field-of-view for a helicopter tandem cockpit

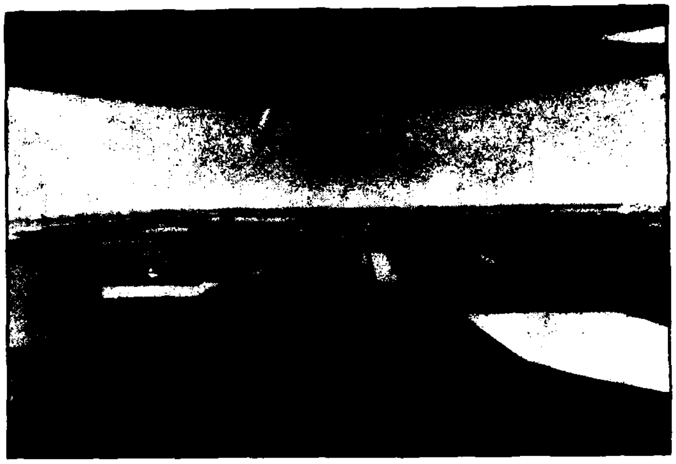


Figure 14 Typical CGI airfield scene

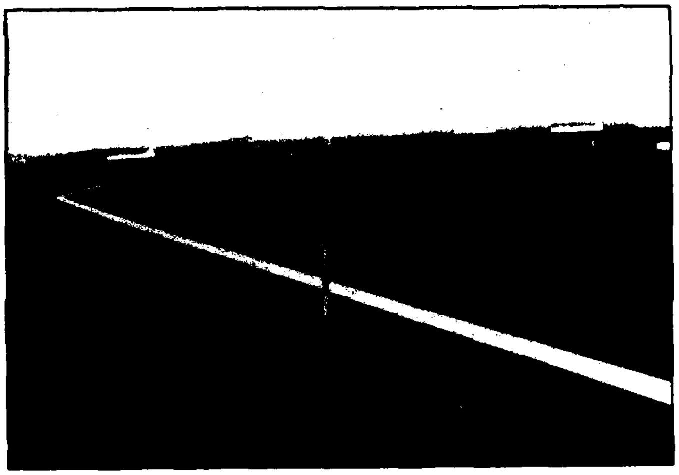


Figure 15 CGI slalom course

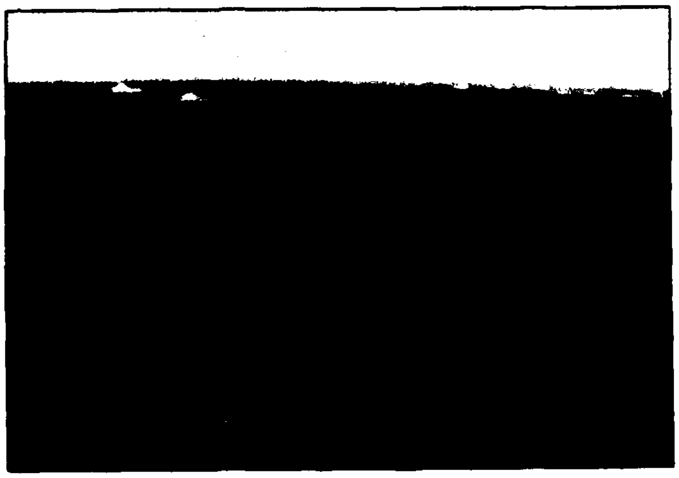


Figure 16 CGI dolphin flight test course

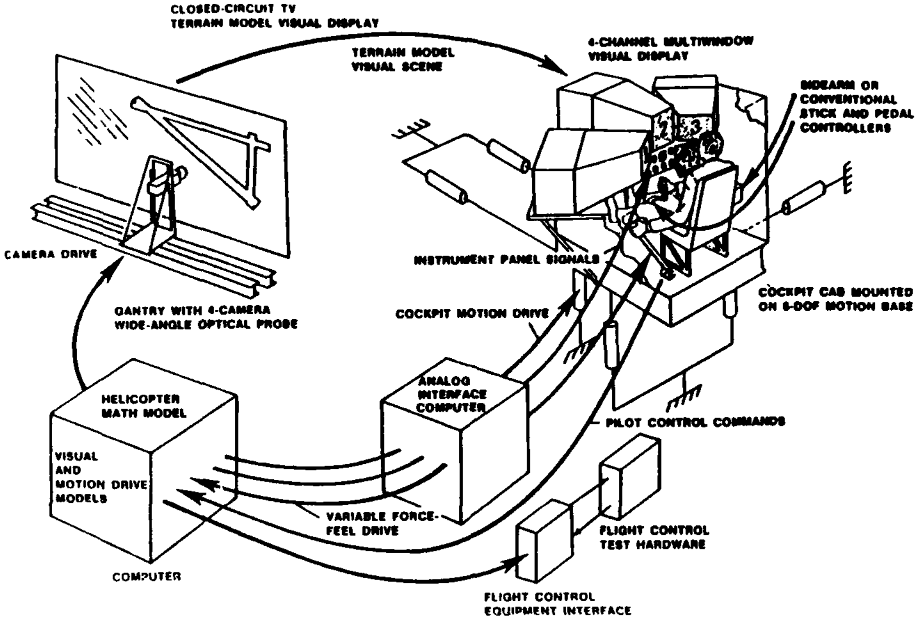


Figure 12 Boeing Vertol flight simulation facility

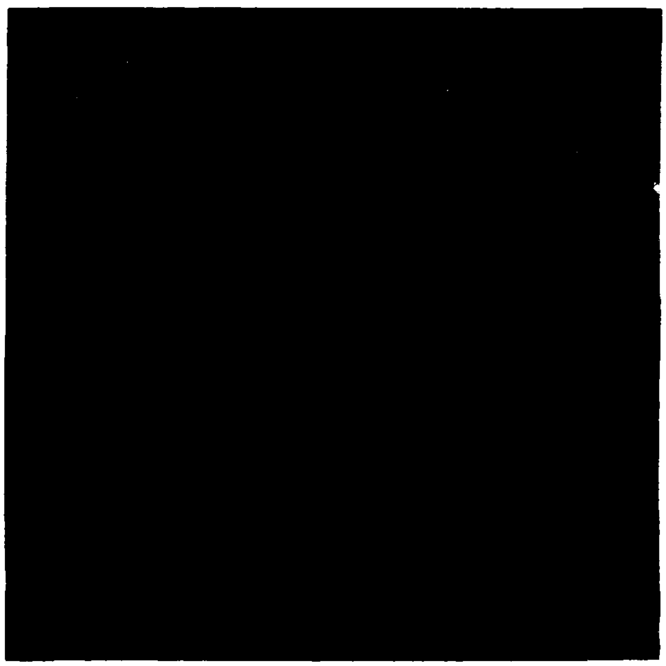


Figure 13 MBB rotorcraft simulation facility,
internal cockpit details and outside CGI-view

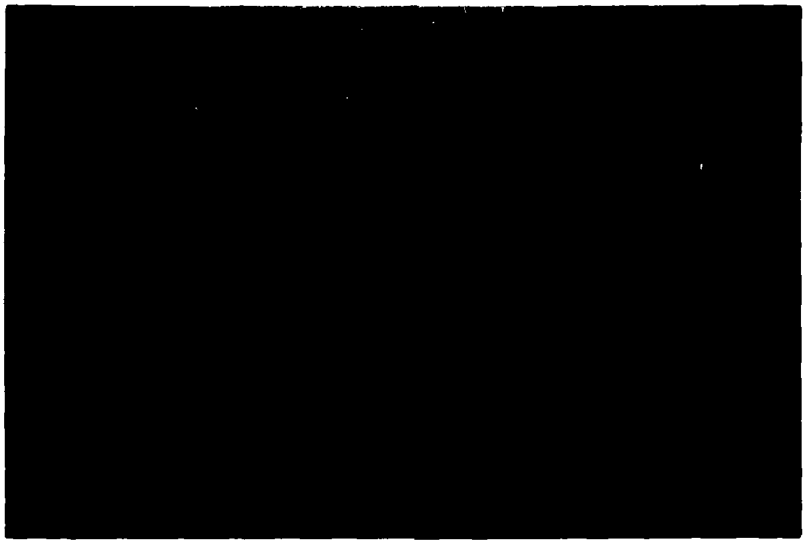


Figure 9 RAE Bedford model board, serpent course, scale 700:1 [16]

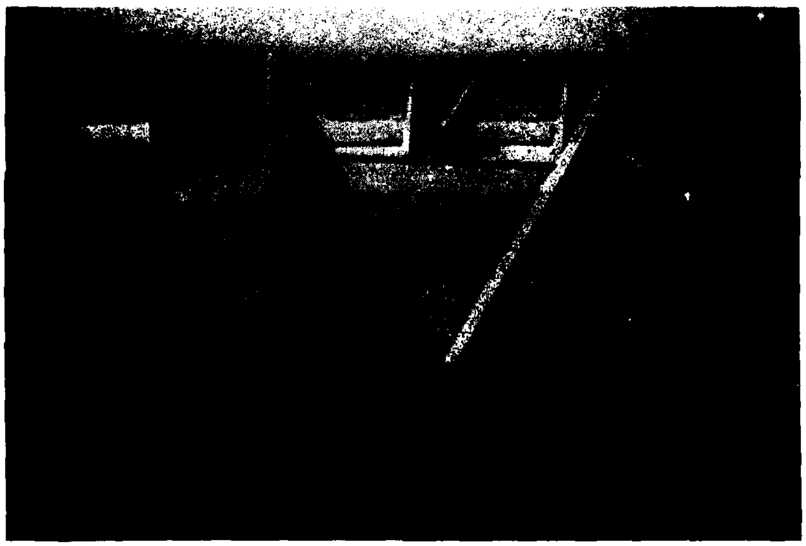


Figure 10 RAE Bedford model board, V-notch hurdles

Figure 11 NASA Ames terrain board detail, scale 400:1 [9]

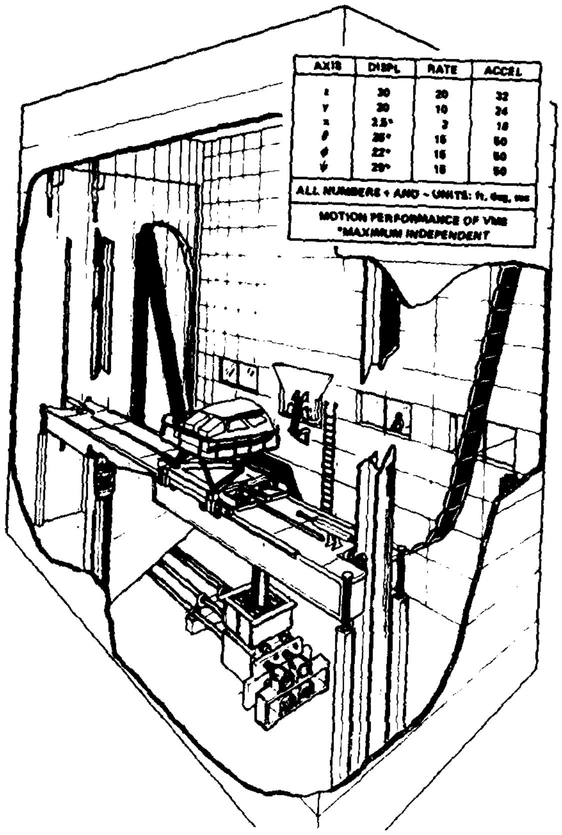


Figure 7 NASA Ames Vertical Motion Simulator (VMS) [10]

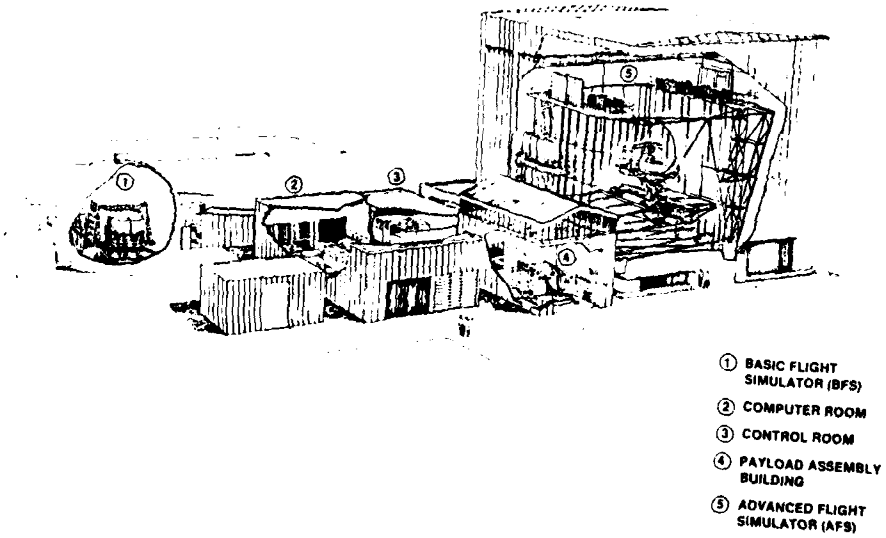


Figure 8 RAE Bedford flight simulator complex [12]

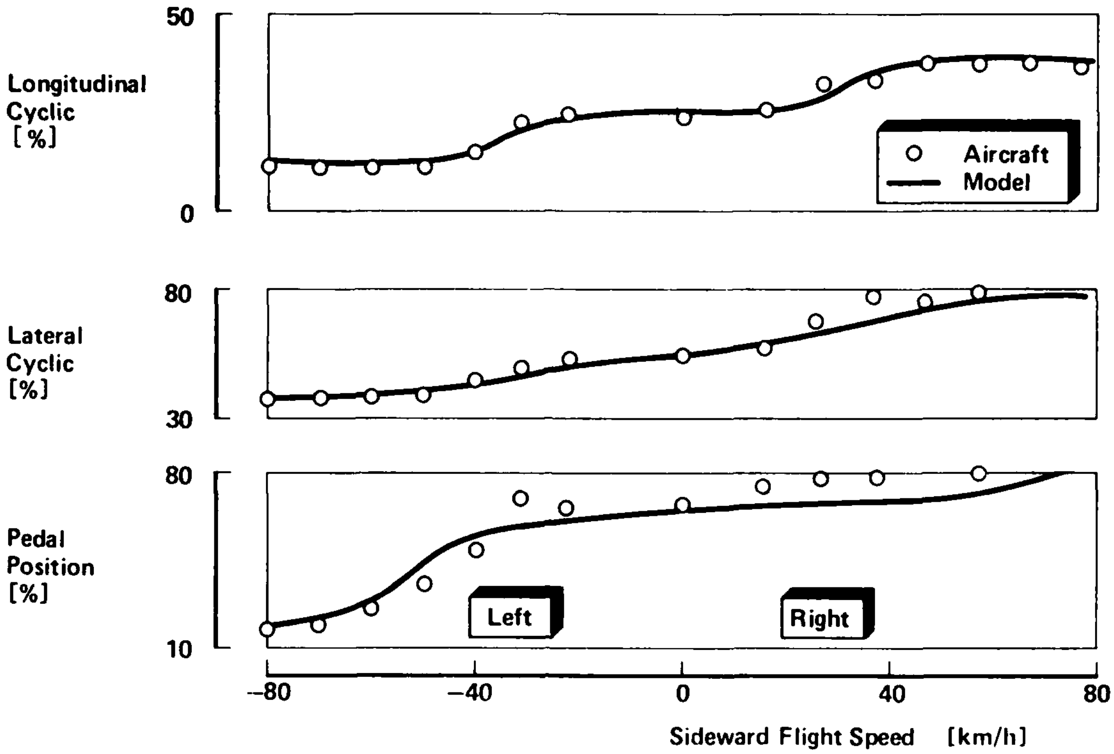


Figure 5 Sideflight trim characteristics

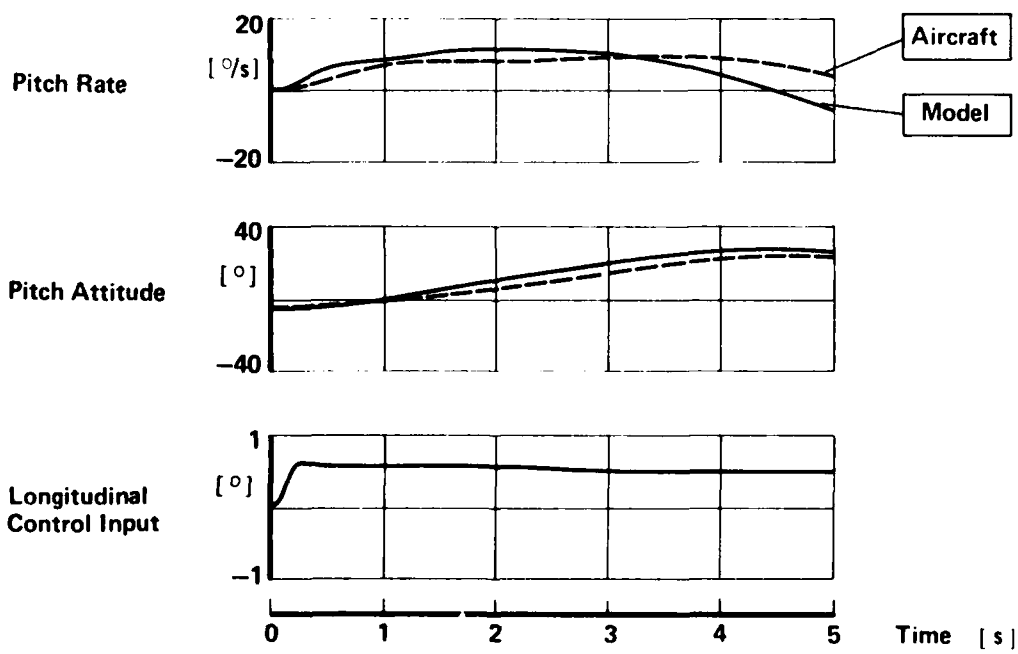


Figure 6 Time histories from flight test and simulation for model validation

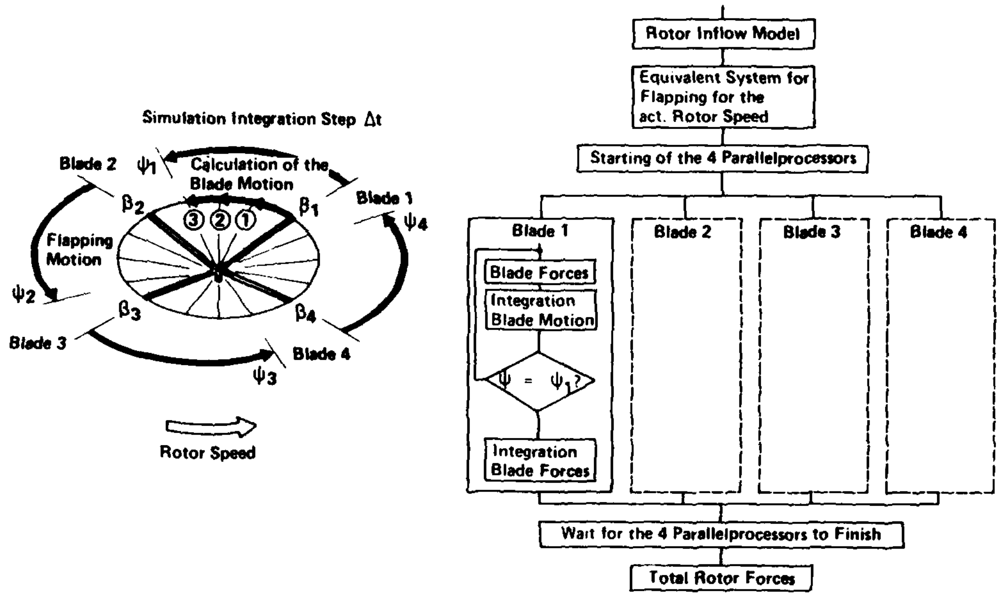


Figure 3 Calculation of blade dynamics

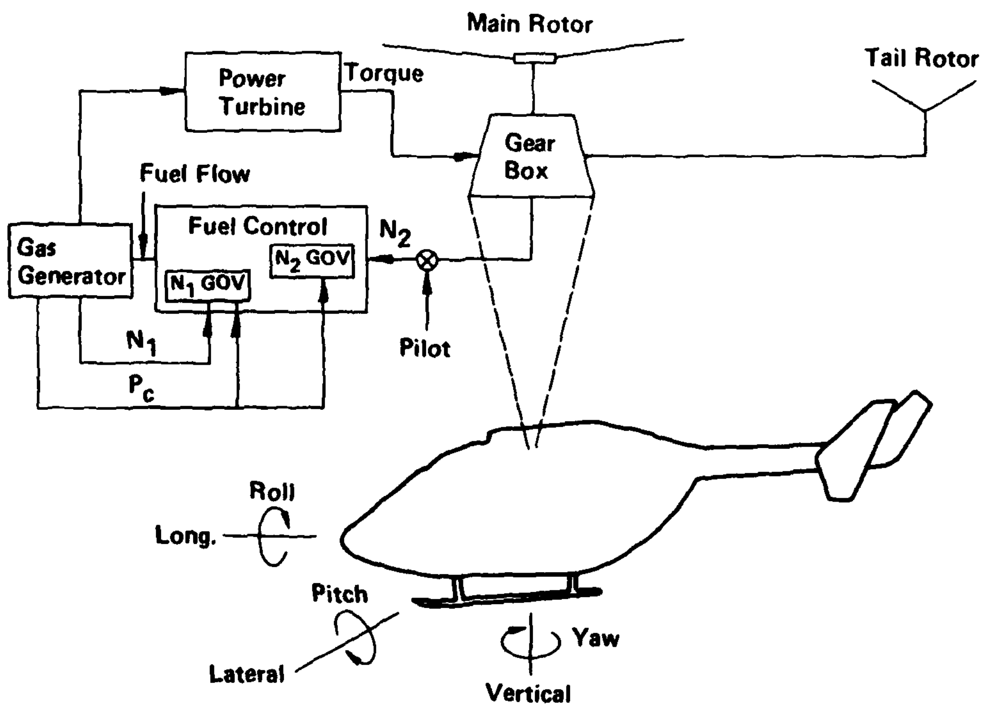
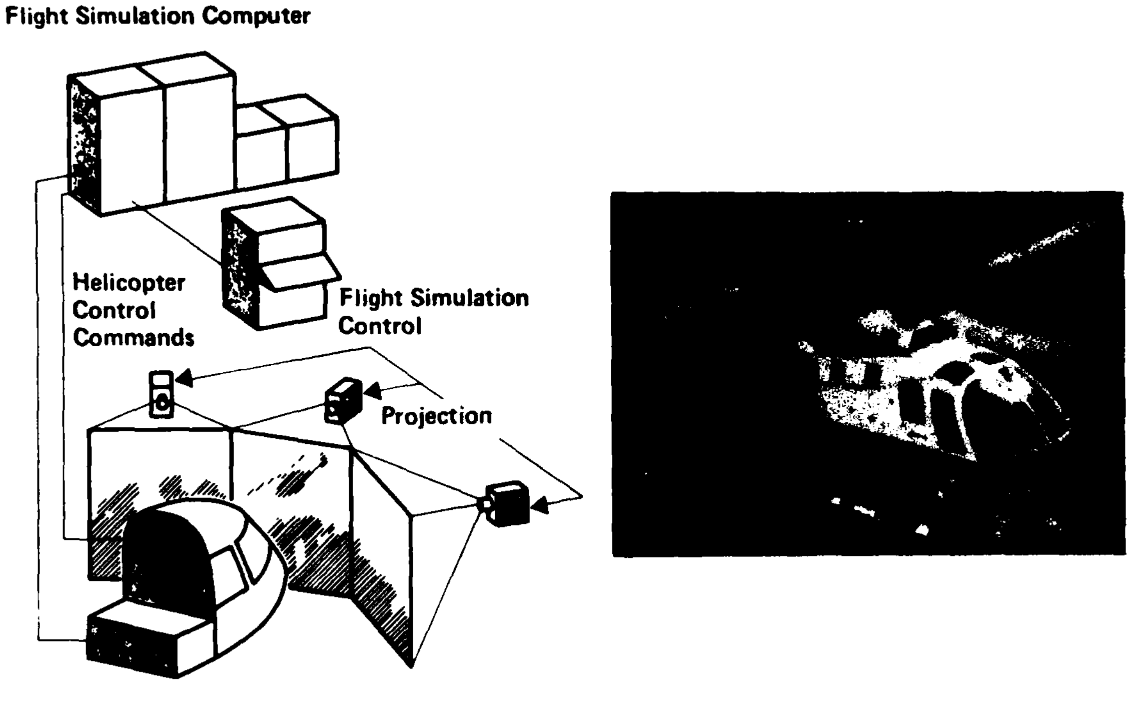


Figure 4 Engine / fuel control modelling



Ground-Based Simulation **In-Flight Simulation**

Figure 1 Different simulation methods

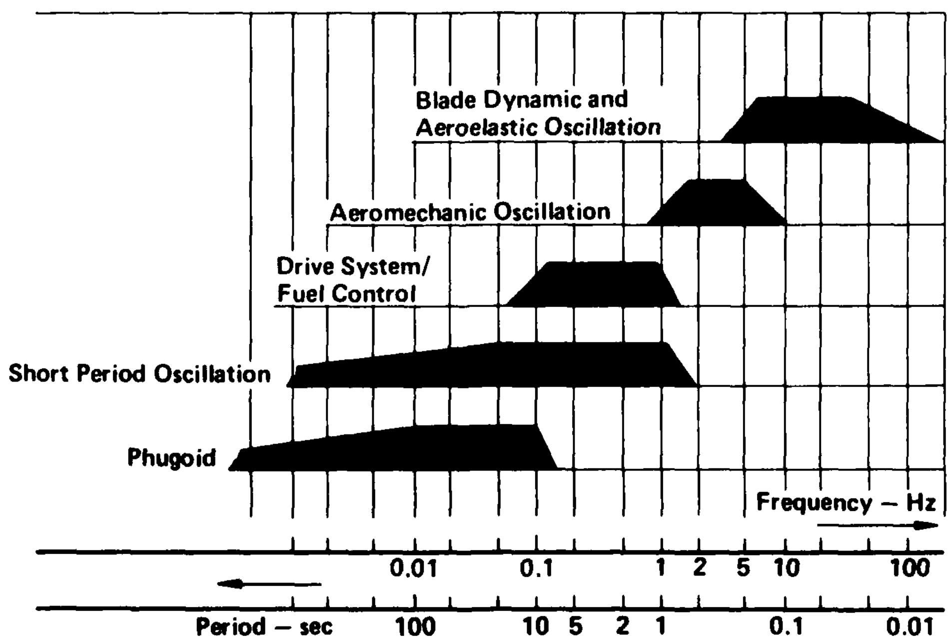


Figure 2 Characteristic frequencies of the helicopter

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The use of ground-based simulation has extended from exploratory handling qualities investigations and is used more and more as a design tool for systems integration and development. Simulation also provides an ideal tool to examine man-machine interface problems and to support verification of soft- and hardware in the loop, well before the first aircraft flies.

Requirements and experience with in-flight simulators were also presented in a short review. It is concluded that airborne simulators can serve in a complementary role to ground-based simulators, especially on those areas where the real-world environment is of particular importance. Future use can be expected mainly for basic investigations and for the evaluation of new flying qualities criteria. However, an intensive use in development efforts will depend on whether such facilities are available in time and can be operated cost-effectively and flexibly.

TABLE 1: ROTORCRAFT MATHEMATICAL MODELS FOR PILOT-IN-THE-LOOP SIMULATION [Ref. 3]

Application	Model complexity*							
	Linear aerodynamics, with simplifications					Nonlinear aerodynamics		
	1	2	3	4	5	1	3	5
General flying qualities - well within flight envelope								
Basic aircraft								
Low-frequency maneuvers	X							
High-frequency maneuvers		X	X					
SCAS research								
Fuselage feedback	X	X						
Fuselage/rotor feedback		X	X	X	X			
General flying qualities - full flight envelope								
Basic aircraft								
Envelope exploration and maneuvering performance						X	X	X
Boundary limiting and expanding SCAS							X	X
Specific aircraft flying qualities								
						X	X	X

*1: Fuselage and quasi-static rotor, 6 DOF.
2: Fuselage and rotor flap, 9 DOF.
3: Fuselage and rotor flap/rpm, 10 DOF.
4: Fuselage and rotor flap/lag, 12 DOF.
5: Fuselage and rotor flap/lag, pitch, rpm, 16 DOF.

TABLE 2: MOTION (PLATFORM) REQUIREMENTS FOR CRITICAL TERRAIN FLIGHT MANEUVERS [Ref. 9]

	Parameter		
Axis	Position, rad, m	Velocity, rad/sec, m/sec	Acceleration, rad/sec^2, m/sec^2
Yaw	±0.4	±0.6	±1.0
Pitch	±0.3	±0.5	±1.0
Roll	±0.3	±0.5	±1.0
Surge	±1.3	±1.3	±3
Sway	±3	±2.6	±3
Heave	+7, -14	+8, -11	+14, -12

Notes: The requirement is for simultaneous operation.
The rotational gimbal order is yaw, pitch, roll.
Translational axes are orthogonal; plus is forward, right, and down.

3.3 IN-FLIGHT SIMULATOR TASKS AND OBJECTIVES

Development of handling qualities criteria. In view of the wide spread field of requirements for basic handling qualities research and rotorcraft development, a large number of in-flight simulator applications is obvious. Perhaps one of the most significant contribution in the recent use were the experiments carried out in support of evaluating flying qualities criteria. The existing handling qualities criteria, especially for military applications, are no longer suitable as design guidelines for the helicopter engineer. The military requirement for Nap-of-the-Earth (NOE) operations has more recently created new needs for flying qualities and agility criteria. Strongly connected with these criteria is the need for including new technologies in displays and controllers.

Evaluation of controls and displays: Due to the constraints which are imposed on nearly all modern aircraft as far as cost and equipment weight are concerned, a compromise has to be found between display and control sophistication. Several studies have shown that the pilot will accept a less complex control system if adequate information is provided (Figure 43 , Ref. 34). The evaluation of these requirements is so difficult that it can only be partially adressed even in the most advanced ground-based simulators. One investigation, carried out on the NAE-205A-1 flying simulator aircraft, was the evaluation of various multi-axis, side-arm controller configurations (Ref. 35). Figure 44 shows a simplified block diagram of a typical simulation channel as used in these series of experiments. The experimental software was arranged so that the various outputs from the handcontrollers enabled a variety of control modes to be investigated.

An important requirement in the design of future pilot controls is the representation of proper control force characteristics and control signal shaping. A corresponding in-flight experimental program has been carried out by MBB in cooperation with DFVLR-Braunschweig to investigate an artificial force-feel system for the control of the pitch and roll axes, using the BO 105-S3 fly-by-wire aircraft (Ref. 36). The goal of the experimental program was to investigate if the use of such artificial force-feel system in pilot controls would allow a more precise flight path control, and reduce pilot workload. Figure 45 presents a schematic outline of the various input terms and their possible combinations, shown for the pitch axis. Most of the actual evaluations were done for two highly dynamic maneuvers, called "dolphin" and "slalom". Resulting from these in-flight experiments, it was found, for example, that damping force contributions from control rates should be suppressed, and that introduction of certain flight dynamic terms are rated highly favourable.

Control and Response and Cross-Coupling: As metioned in a foregoing section, important considerations that can strongly influence the handling qualities are the control response and cross-coupling characteristics. There is a lack of data for specification of control system response types, and the optimum hierarchy of control and stability augmentation. Control and vehicle cross-coupling characteristics have further fundamental influence on pilot workload and task performance. In-flight simulation is particularly suited to verify those data which are available from ground-based simulation.

One further important area of these efforts is the determination of boundaries, which define minimum acceptable standards for FAA airworthiness criteria, or minimum (level II and III) military flying qualities criteria (Ref. 32). There is also a lack of data to specify acceptable levels on degraded handling qualities, in case of partial or complete failure of control systems or vision aids. In-flight simulation is one of the most realistic tool to support development of new criteria on all the above areas.

3.4 SUPPORT OF DESIGN AND DEVELOPMENT EFFORTS

Application of in-flight simulators as development tool for particular designs was limited until recently due to the limited number and the limited capabilities of in-flight simulator aircraft. However, existing airborne-simulators are being improved, and new variable-stability aircraft are in various stages of development; hence, it can be anticipated that the interest will increase gradually, to use these aircraft to support engineering and man-machine interface efforts also during the design and development phase. Key contributions can especially expected in the fields of (Ref. 30) basic parameter investigations, pre-production verfication, flight control system development, hardware in-flight testing, simulation of system failures, and in supporting efforts during certification procedures.

From the above it can be concluded that a clear need for future helicopter airborne-simulation is seen. Still, it should also be mentioned that some responses, especially from industries, show a more reserved character. The opinions are ranging from too high building cost and problematic cost-effective operation and some reservations are also made with respect to availability and control over such a simulation facility (Ref. 2).

4. CONCLUSION

It was the aim of this lecture to give an overview about the various simulation techniques and their specific applications during research and system development work of rotorcraft. It should be noted that the role of simulation is, of course, a theme of high complexity and challenge, and it is impossible to treat it completely in the scope of a survey like this. It was intended to select and present some of the main elements and requirements for adequate simulator fidelity in order to simulate the missions and tasks that are unique to rotary-wing aircraft. Aspects of mathematical modeling were reviewed and the importance of comprehensive helicopter math models, showing high dynamic fidelity up to the boundaries of the flight envelope was discussed. Although it has not been discussed here in detail, motion system fidelity requirements for ground-based simulators were also briefly reviewed.

Visual display requirements in particular for military mission simulation up to now were the most critical limitations on environment simulation. However, the technology of computer Generated Imagery (CGI) has advanced to the point that most of helicopter terrain flying tasks can be simulated with a high degree of fidelity. The technology is still rapidly advancing in this area, offering most promising capabilities in the near future.

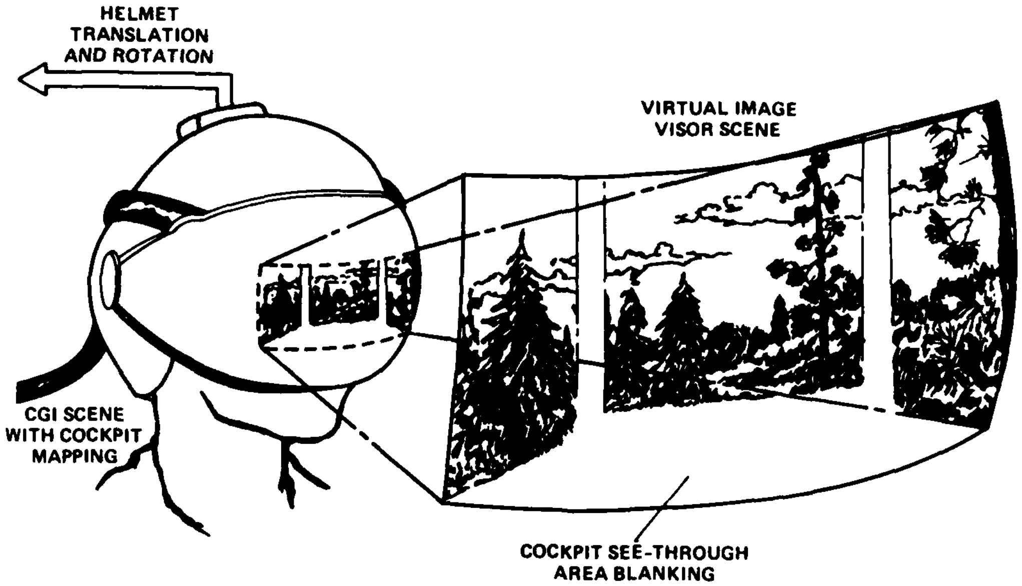


Figure 23 Artist's conception of helmet-mounted visor display blanking [10]

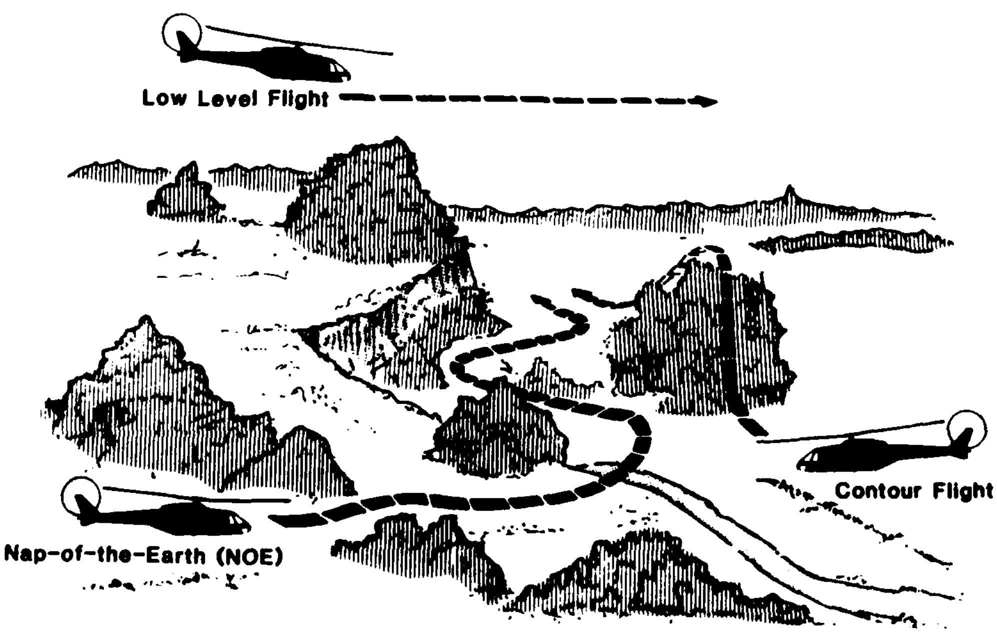


Figure 24 Typical helicopter mission flight profiles

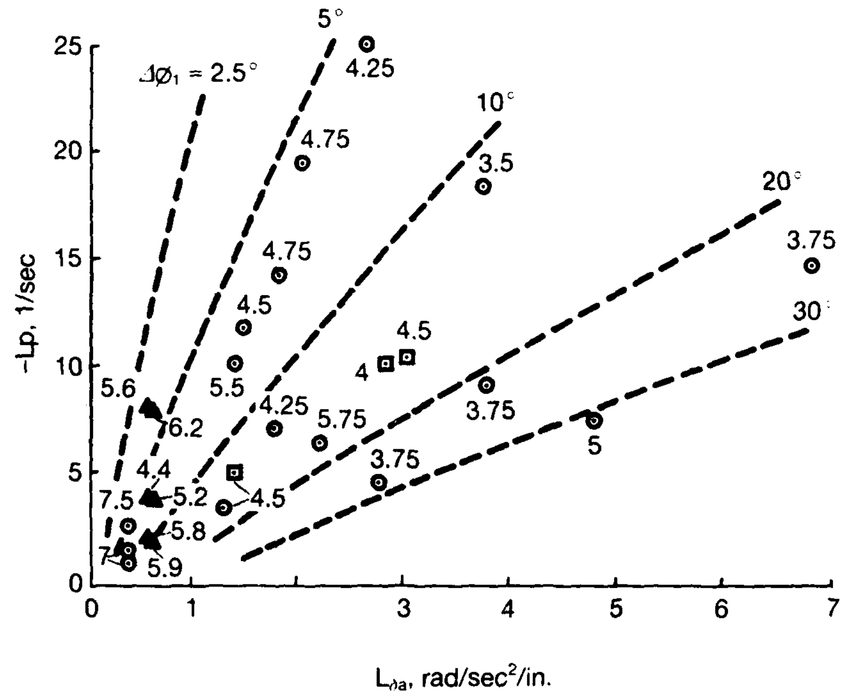


Figure 25 Effect of roll damping and sensitivity on average pilot rating [21]

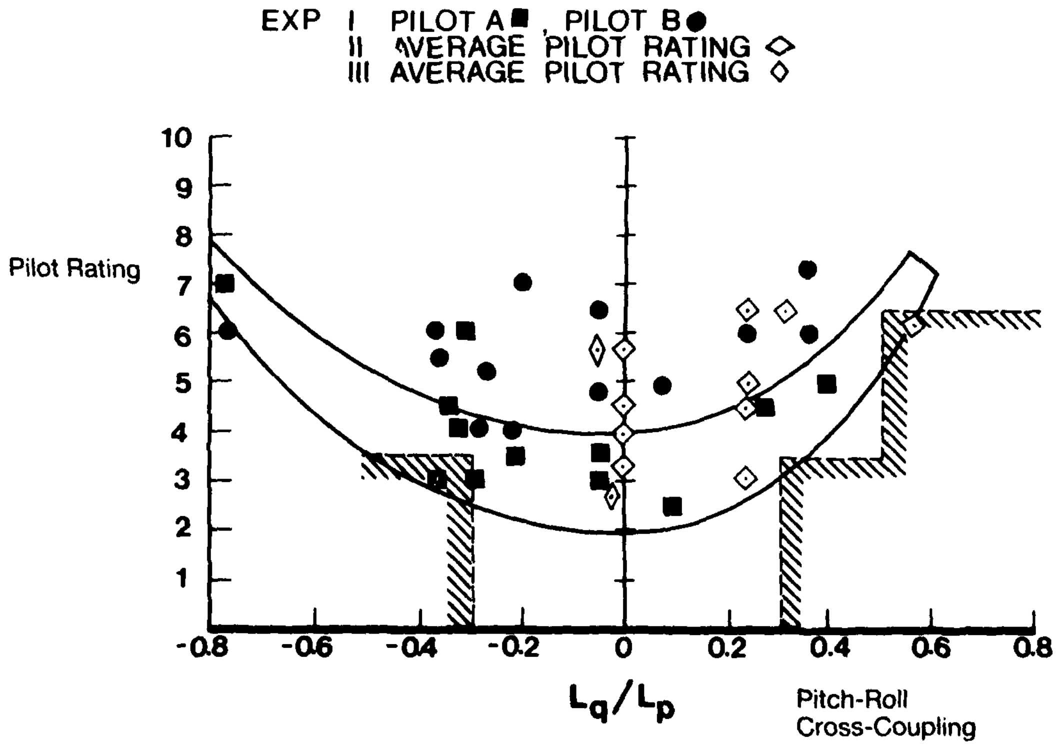


Figure 26 Pilot rating vs. pitch / roll cross coupling (combination task) [21]

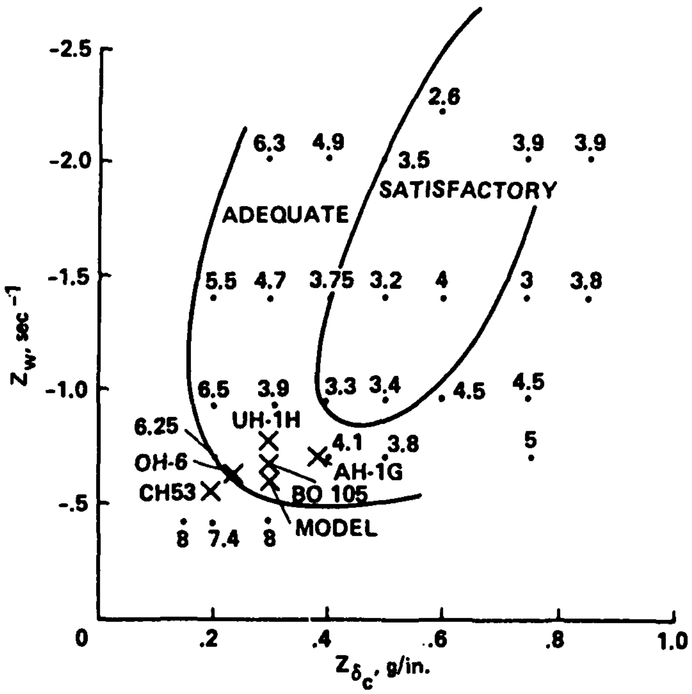
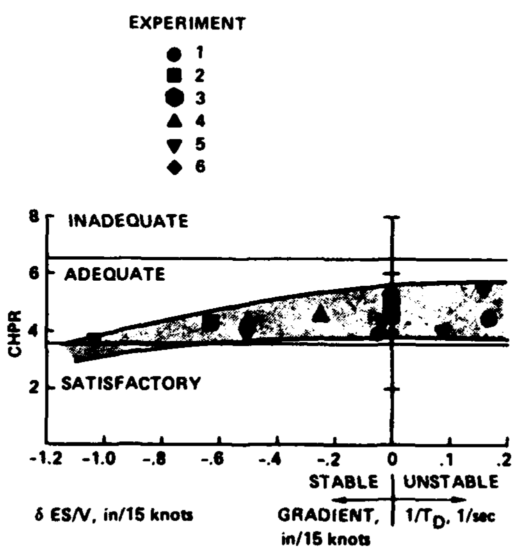
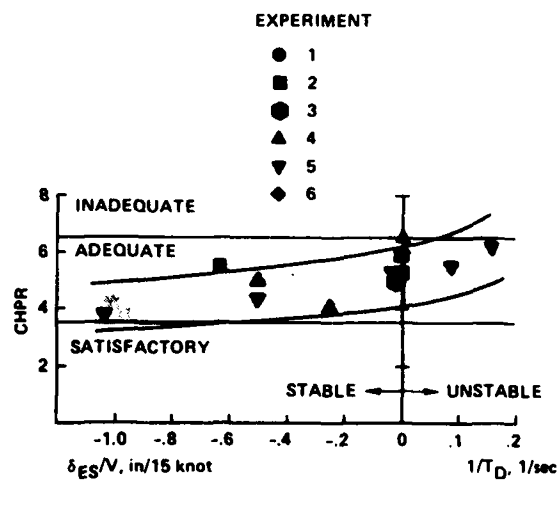


Figure 27 Vertical damping and collective sensitivity (dolphin maneuver) [22]



(a) No turbulence, no flight directors.



(b) In turbulence, no flight directors.

Figure 28 Pilot rating data as a function of longitudinal stick gradient [23]

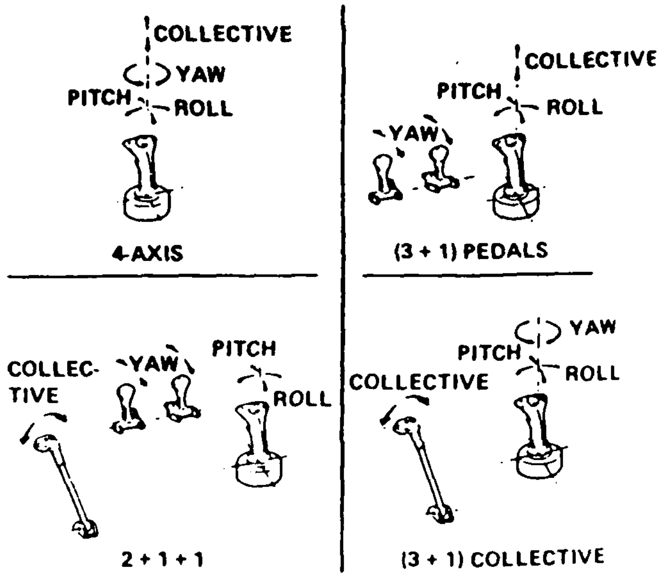


Figure 29 Various side-stick controller configurations [17]

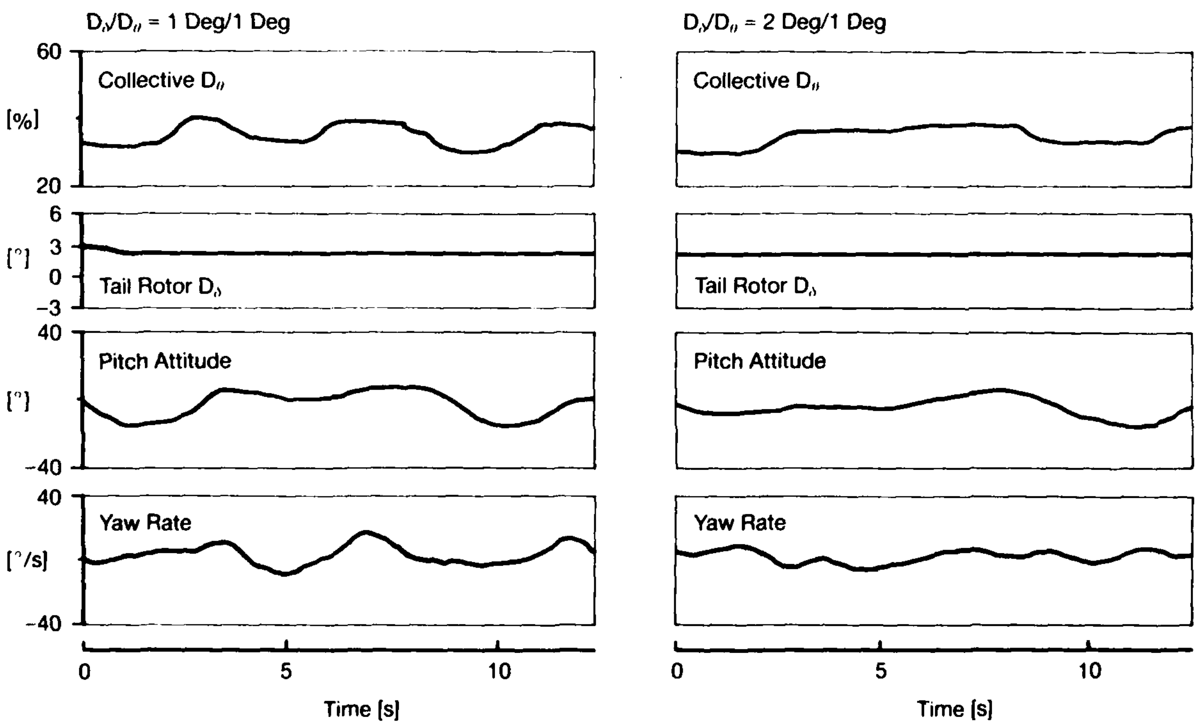


Figure 30 Effects of collective / tail rotor cross coupling on dolphin maneuver

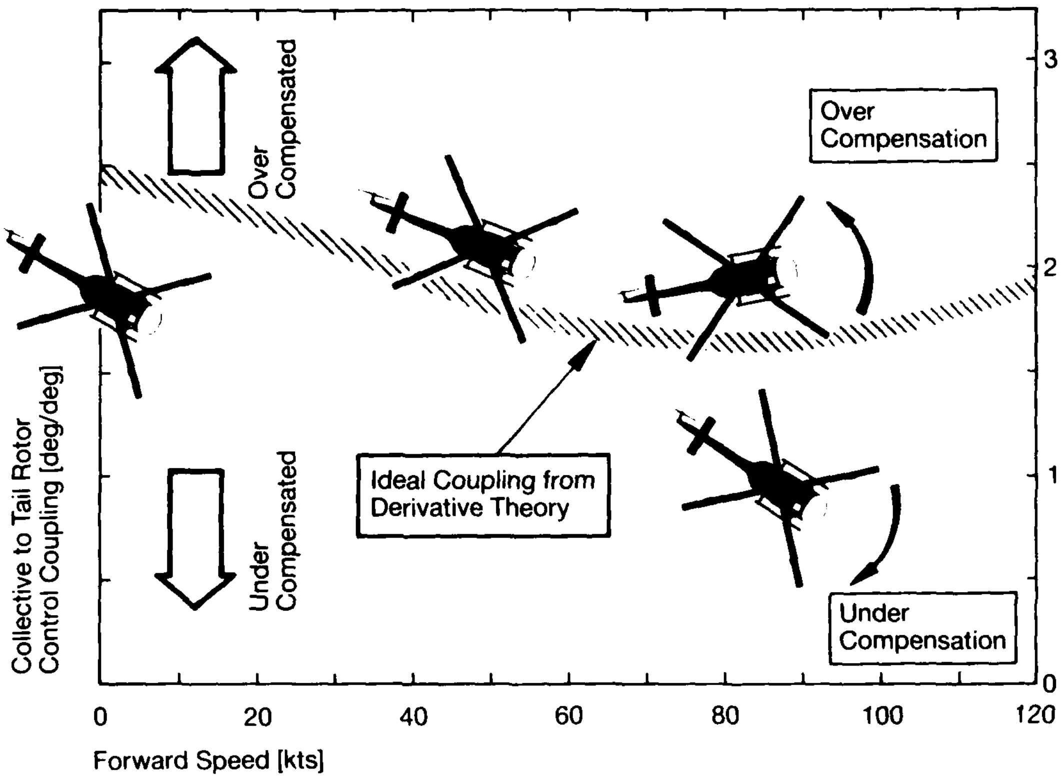


Figure 31 Comparison of derivative and simulator results of collective to yaw control coupling

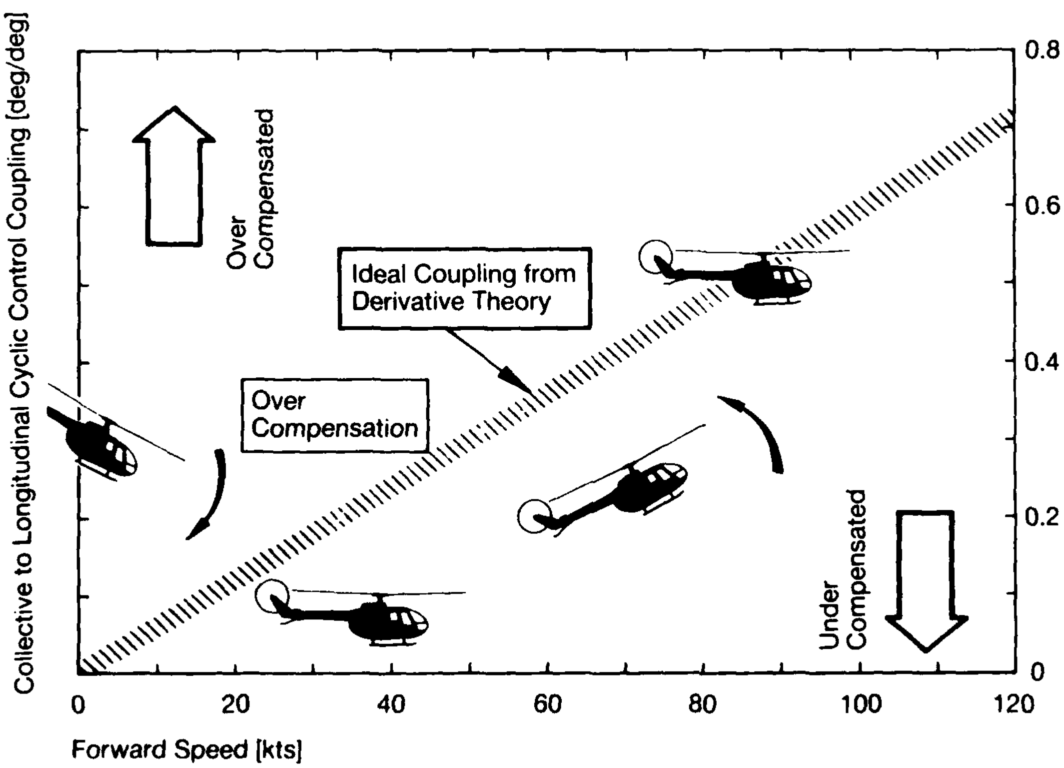


Figure 32 Comparison of derivative and simulator results of collective to longitudinal control coupling

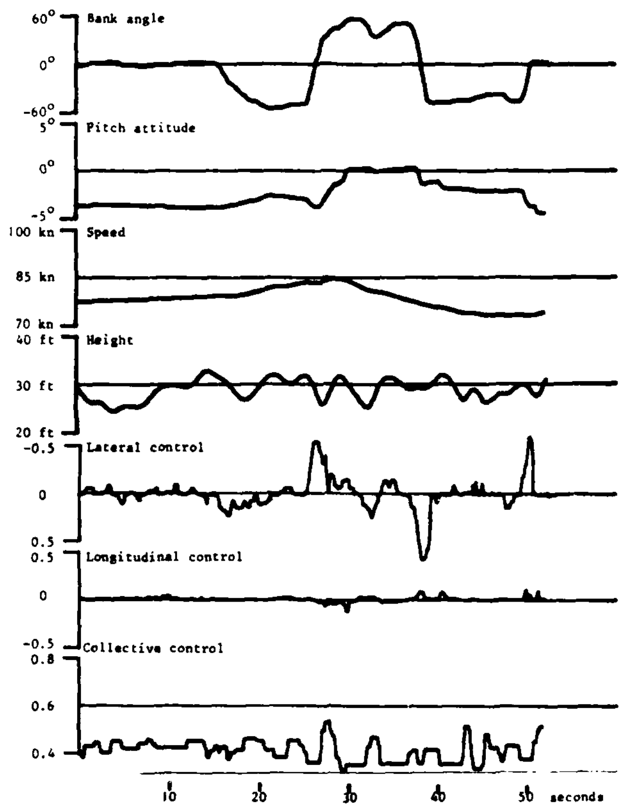


Figure 33 Time histories of simulated serpent course flight (full augmentation) [16]

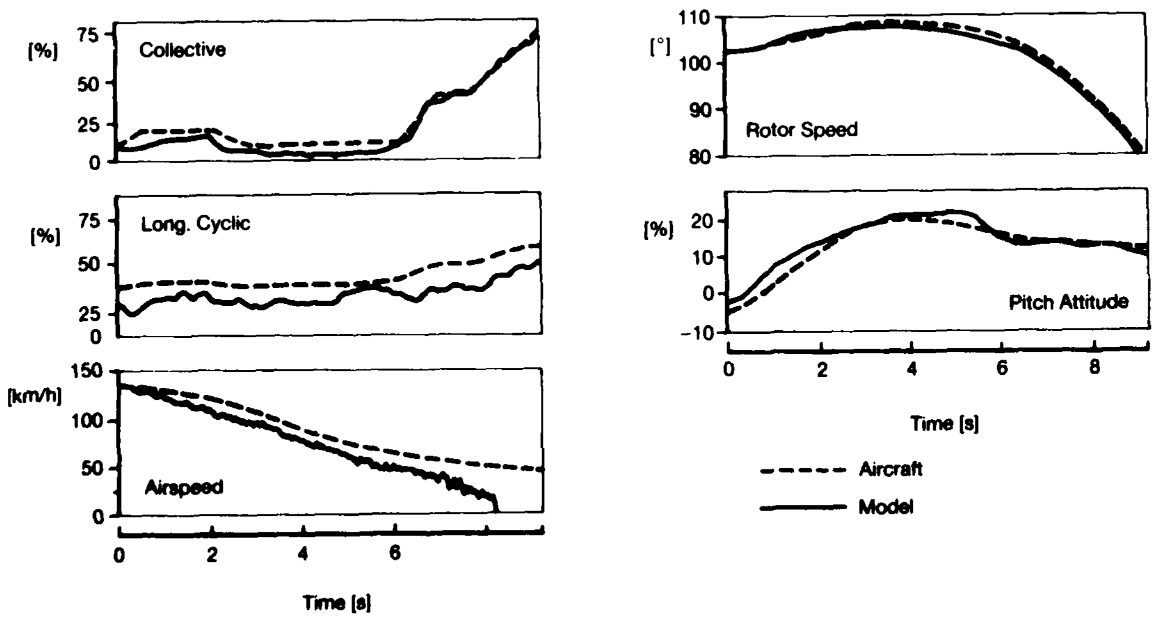


Figure 34 Time histories of simulated engine failure

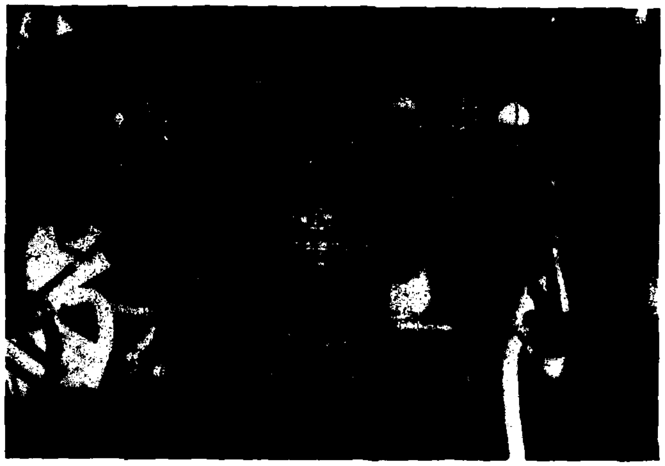


Figure 35 Current generation helicopter cockpit interior

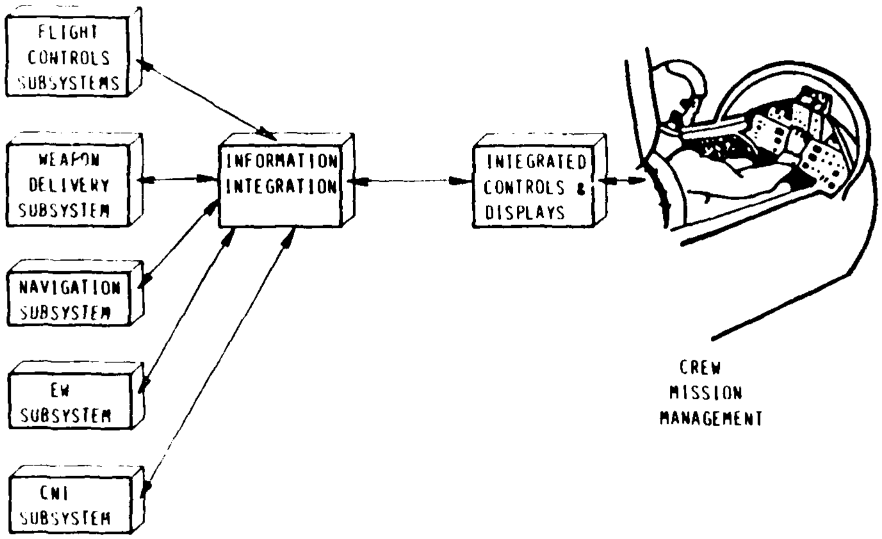


Figure 36 Highly integrated coordinated aircraft systems [39]

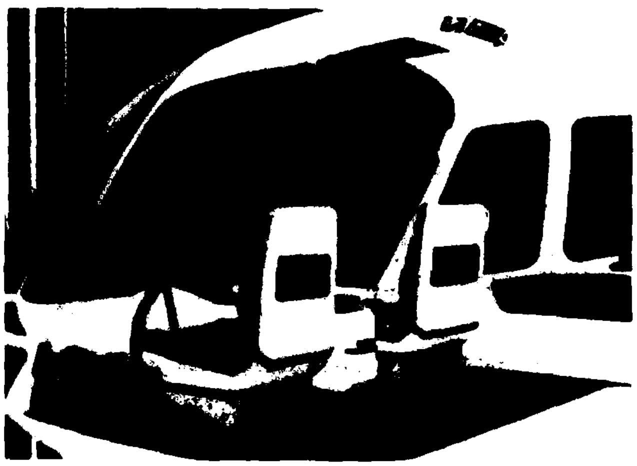


Figure 37 Light civil helicopter cockpit mockup

Figure 38 Pilot's night vision goggles

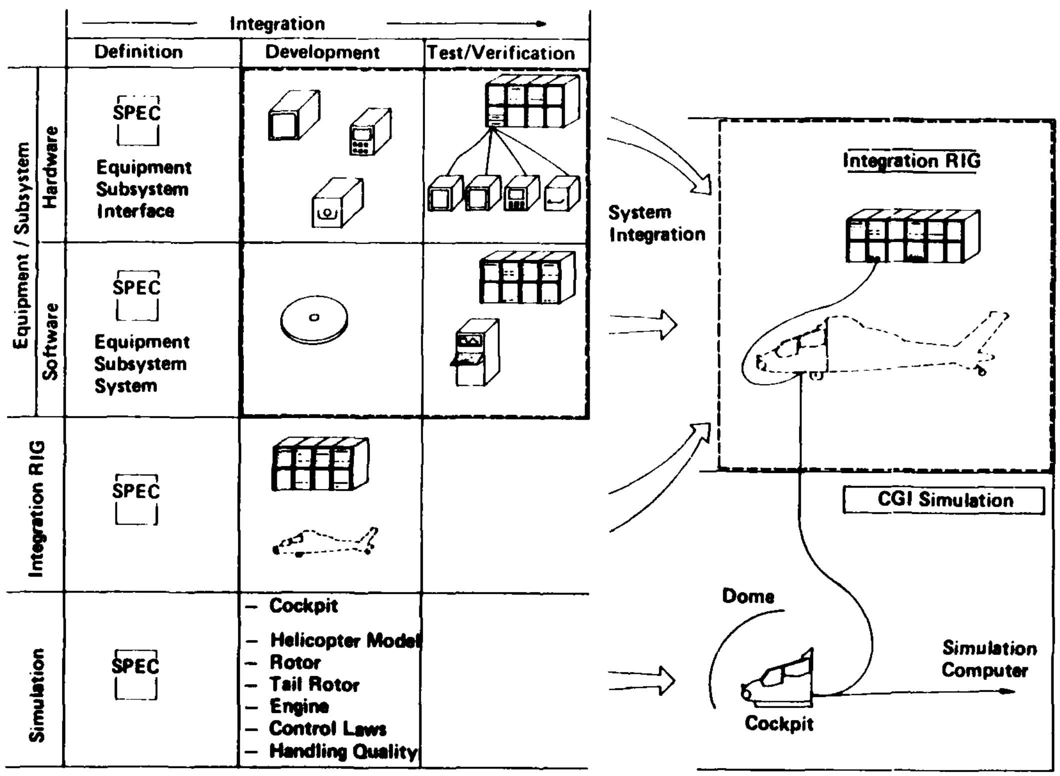
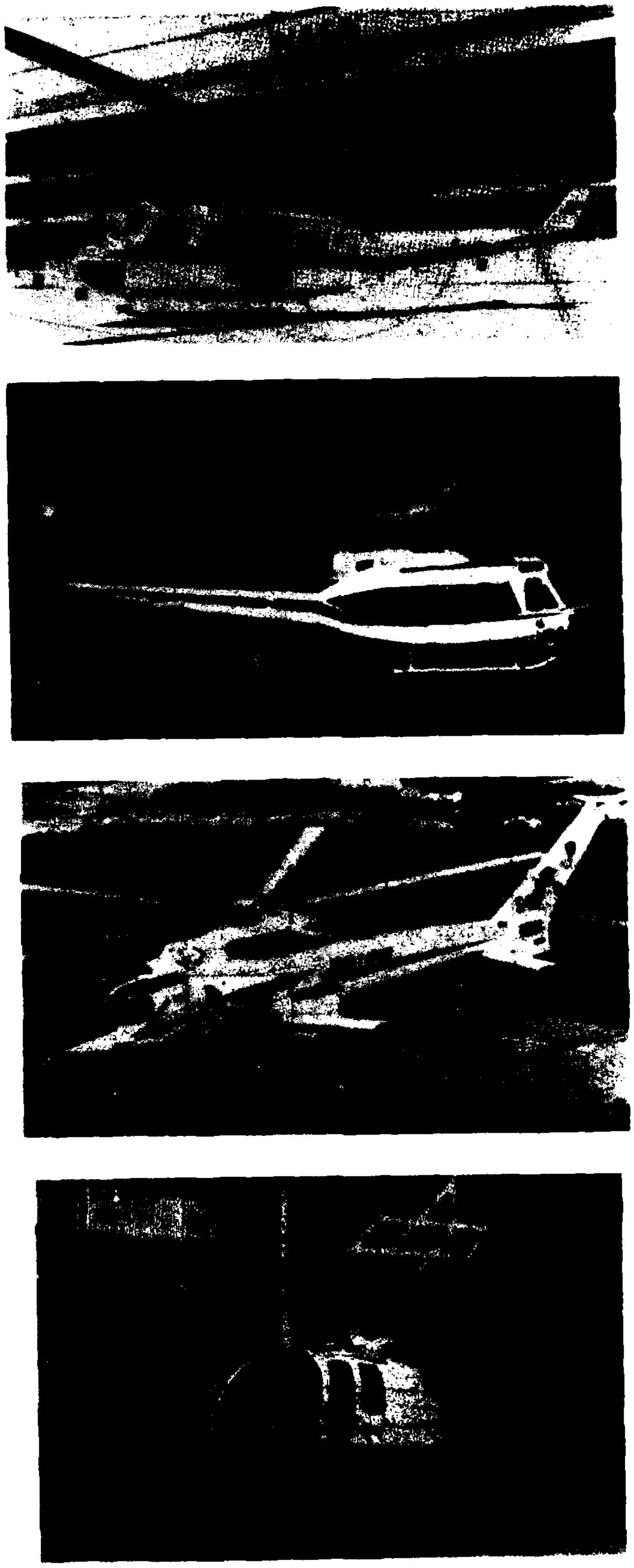


Figure 39 System integration concept



From top to bottom:

NASA / Army UH-1H (V/STOLAND)
NAE Bell 205A-1
NASA / Army RSRA
DFVLR BO 105 ATTHeS

Figure 40 Existing in-flight simulators

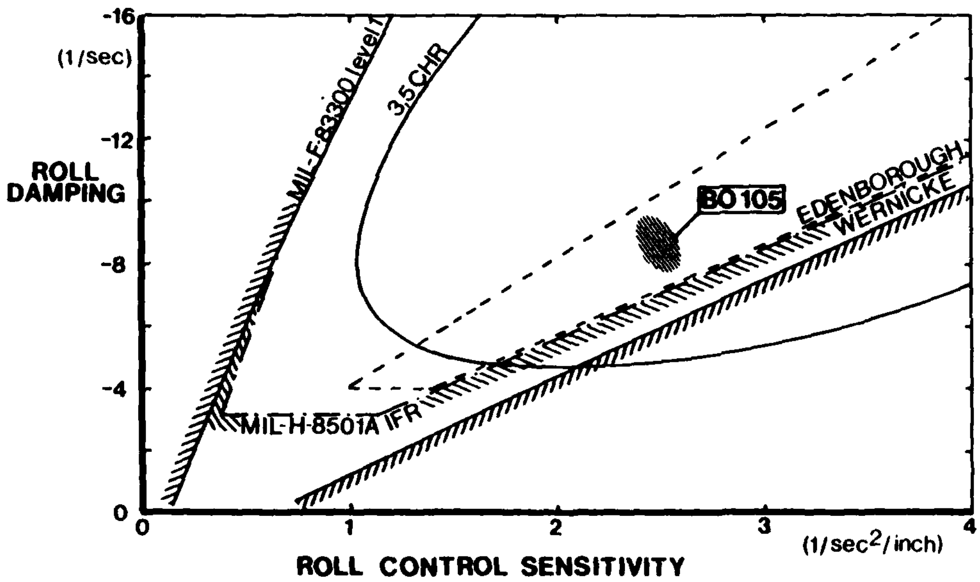
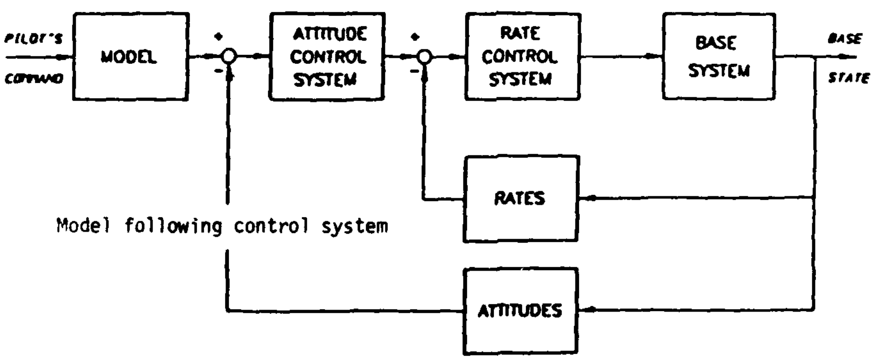


Figure 41 Lateral control response characteristics [40]



Model following control system

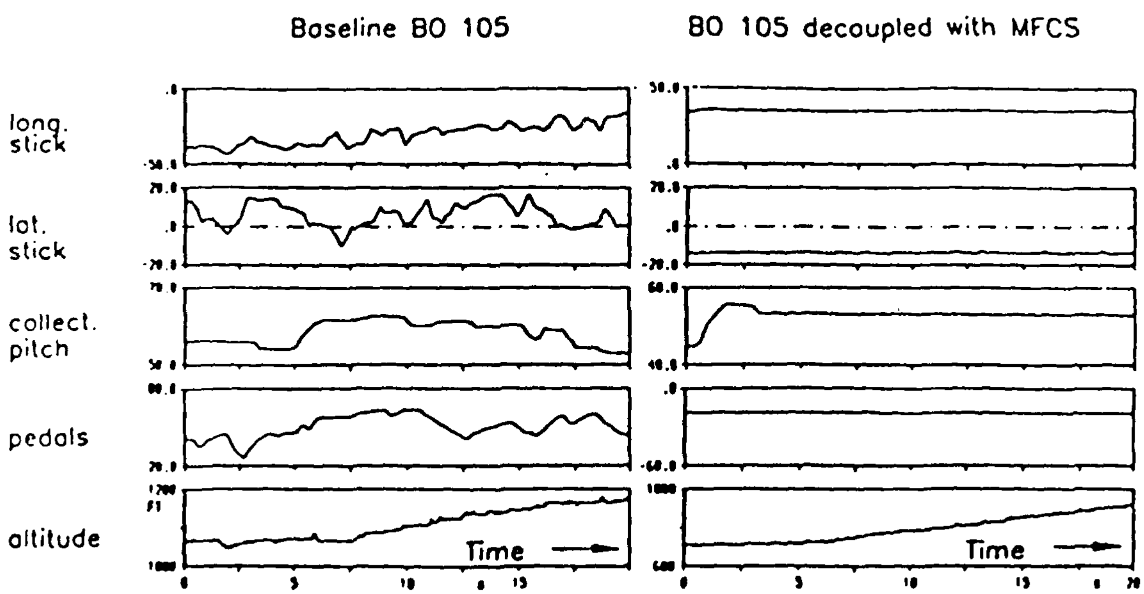


Figure 42 Comparison on pilot's control activity in climb,
without and with model following control system (MFCS) [30]

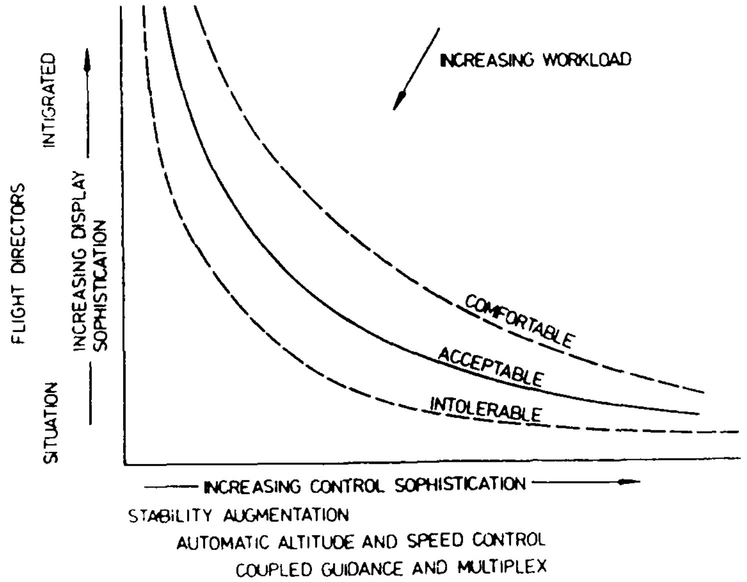


Figure 43 Interdependance between display complexity and control automation [36]

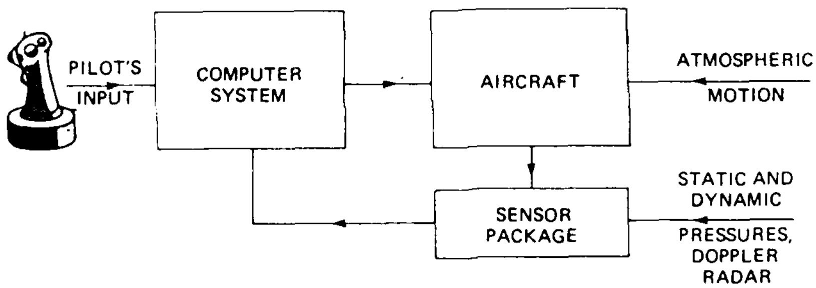


Figure 44 Typical simulation channel for side-arm controller experiments [35]

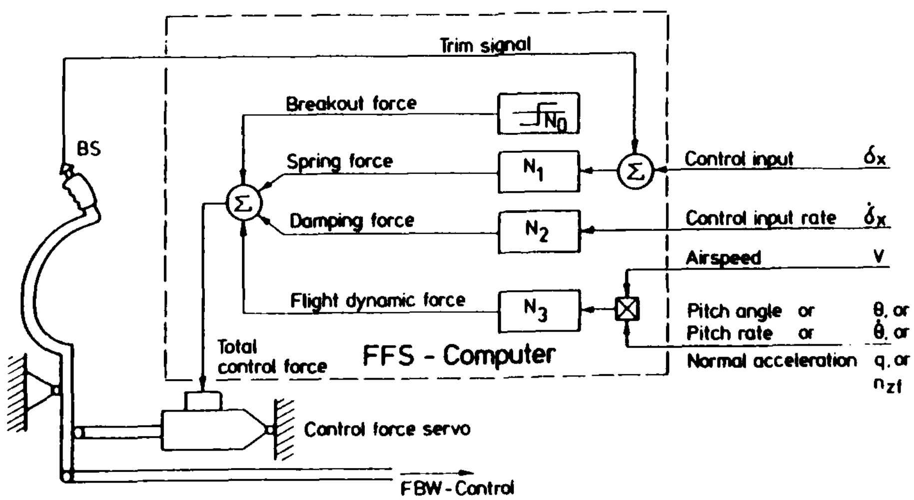


Figure 45 Control force contributions for the pitch axis [36]

FLIGHT TESTING FOR PERFORMANCE AND FLYING QUALITIES

by

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SUMMARY

This lecture provides a systematic review of flight test techniques and test data interpretation methods for helicopter performance and flying qualities. The distinction is drawn between quasi-steady and dynamic testing and within these categories both clinical and role-related techniques are discussed. Performance topics covered include steady state performance in hover and forward flight, flight envelope boundaries, take-off and landing performance, and helicopter agility. Flying qualities topics begin with a treatment of static stability tests and progress to dynamic stability, control response, system identification and role-related evaluation techniques. Testing appropriate to certification and development phases and research activities are addressed. The exploratory nature of flight testing is evident throughout this work and safety aspects are emphasised when required. Results from recent and past test programmes are used to illustrate the forms in which flight data can be presented, and data reduction and analysis methods established and under development are reviewed.

LIST OF SYMBOLS

a_o	reference speed of sound
g	acceleration due to gravity
h	height above ground
L, M, N	aircraft roll, pitch and yaw moments
L_p, M_q etc	moment derivatives normalised by inertias
$\bar{M}_t$	mean tip Mach number
n	load factor; rotorspeed ratio
P	power input
p, q, r	roll, pitch and yaw rates
P_c	power coefficient
Q	torque
R	rotor radius
R_E	effective radius of turn
s	rotor solidity
T	rotor thrust
TAF	turn agility factor
t_c	rotor thrust coefficient
U_o	trim value of velocity component along aircraft fuselage axis
u, v, w	aircraft velocity components
V	flight speed
V_c	vertical climb rate
V_{mp}	flight speed for minimum power
V_{ne}	never exceed flight speed
W	aircraft weight
Z	force component normal to aircraft fuselage axis
Z_w, $Z_{\theta 1s}$	Z-force derivatives normalised by mass
δ	atmospheric pressure ratio
ζ	damping
η_{1c}	lateral cyclic control position
η_{1s}	longitudinal cyclic control position
η_p	pedal control position
θ	atmospheric temperature ratio
θ_{1c}	lateral cyclic pitch angle
θ_{1s}	longitudinal cyclic pitch angle
μ	advance ratio
ρ	atmospheric density
σ	atmospheric density ratio
Ω	rotor speed
ω	rotor speed ratio

ABBREVIATIONS

AAH	advanced attack helicopter
AFCS	automatic flight control system
ASI	airspeed indicator
AUW	all up weight
agl	above ground level
CDP	critical decision point
cg	centre of gravity
EAS	estimated airspeed
FAA	Federal Aviation Authority
Fig	Flight Idle Glide
FFT	fast Fourier transform
IAS	indicated airspeed
IFR	instrument flight rules
IGE	in ground effect
ISA	international standard atmosphere
LAMPS	light airborne multi-purpose system
NOE	nap-of-the-earth
NIPIP	non-intrusive pilot identification procedure
OAT	outside air temperature
ODM	operating data manual
OGE	out of ground effect
PIO	pilot induced oscillation
ROC	rate of climb
ROD	rate of descent
RTO	rejected take-off
SAS	stability augmentation system
SCAS	stability and control augmentation system
SIFT	system identification from tracking
TAF	turn agility factor
UCE	usable cue environment
UTTAS	utility tactical transport aircraft system
VFR	visual flight rules
VTOSS	take-off safety speed

1 GENERAL INTRODUCTION

Flight testing allows the ultimate assessment of a new concept after an involved evolutionary process from initial ideas, through analysis and simulation, to engineering design, model scale and ground rig tests. The level of risk involved in flight testing, from both safety and cost viewpoints, depends upon the confidence gained and the successes achieved during this evolution. Most ideas, largely for cost reasons, only reach the analysis stage and of those that progress further, only a small percentage will be awarded the privilege of undergoing the ultimate trial. For this reason, flight testing is required to be a very professional discipline and test techniques, measurement equipment and safety philosophy need to be carefully defined if valid and repeatable results are to be obtained. The flight tests required to define the performance and flying qualities of a new or modified helicopter are particularly demanding in this context; experience over the twelve years since the previous AGARD lecture series on this topic (Ref 1) with more than a dozen new helicopter designs entering operational service, has underscored not only the importance of flight test development but also that re-design during this late phase can be very costly to a programme. The technological improvements in the next generation of helicopters, being developed to meet more demanding requirements and standards, emphasise the need to disseminate the knowledge base generated by experience as widely as possible, hence minimising the risk for those privileged ideas that reach flight test.

The various categories of flight testing include those conducted during the Manufacturer's development programme, the Government test agencies compliance demonstration programme and research testing. In some cases, if an aircraft is being built to a particular military specification the development and compliance demonstration programmes benefit from being integrated, hence avoiding unnecessary duplication of new test facilities as well as the tests themselves. This integrated approach is already in practice in some countries. Government testing will include that required to produce data for the operating data manual and pilot's handbook. In addition to clinical tests to gather specific data, techniques are becoming established under the category of role-related testing which will often include an operational evaluation with a Service Squadron.

This lecture is concerned largely with test techniques and as such is applicable to all three categories referred to above. Continued reference is made to the kind of data needed to satisfy requirements. Within the framework of these different categories there are a number of additional perspectives on flight testing. The airframe and system designers will be interested in how well the results match their predictions, how to interpret the results and what re-design effort is required. The test pilot and flight test team will be concerned with coordinating and planning the trials, deriving new and more efficient ways of collecting data, keeping the aircraft and instrumentation serviceable and will, at least at some test sites, be at the mercy of the weather. The qualification agency will be comparing test results with military or civil requirements and the potential operator will be looking for data on performance and flight limitations.

The researcher, though his perspective may be more limited in scope, will be more interested in the fine details in the measurements and developing ways of converting these into useful information.

In writing this lecture the author has attempted to portray a blend of these varied perspectives. To accomplish this task, material has been gathered from many different sources including the UK and US Military and Civil Requirements (Refs 2-7), Test Pilot's training manuals (Refs 8-10), the US Army Engineering Design Handbooks (Refs 11-12) and various published papers on test experience with different helicopter types. Acknowledgement of these sources is included at the end of this lecture.

The two main chapters deal with Performance and Flying Qualities testing. Within these, there are sub-sections on quasi-steady testing and dynamic testing. Both clinical and role-related test techniques are described and examples given from recent and past programmes.

2 PERFORMANCE TESTING

2.1 Introduction

Performance flight testing is concerned chiefly with two principal objectives; namely the verification of an aircraft's actual performance in both steady and manoeuvring flight and the definition of the flight envelope limitations and how these can be respected. Operating limitations that define the flight envelope boundaries are not, of course, based only on simple power considerations but include airspeed and manoeuvring structural limitations, transmission torque limits, rotorspeed limits and handling deficiencies. The last of these aspects will be addressed in Chapter 3. Tests within the safe flight envelope will normally verify hover and forward flight performance, climb and descent performance and endurance and range characteristics. These quasi-steady performance tests are treated in 2.2. Testing that is aimmed at establishing whether an aircraft has the performance to transit safely from one flight state to another will include take off and landing performance, task orientated manoeuvres and recovery from flight critical conditions *eg* vortex ring. These and related dynamic performance tests are discussed in 2.3.

There are several deliberate omissions from this chapter on performance testing. Firstly, it is considered outside the scope of the lecture to address the important area of flight testing to validate powerplant performance or the related areas concerned with engine and rotorspeed governing. Secondly, testing appropriate to single main rotor helicopters is focussed on, although much of the testing would be equally appropriate to tandem configurations. What are not dealt with are the special techniques that are clearly required for advanced compound or 'convertable' rotorcraft. The third area not covered concerns the expanding number of test programmes in Europe and the US that evaluate the performance benefits that accrue from an airframe design change by making appropriate modifications to an existing test vehicle. Examples include the extensive flight test research at RAE using Wessex and Puma test vehicles to evaluate the detailed aerodynamic characteristics of new aerofoil sections. The new test sections have been built around the standard blade profile over a limited radial extent, and miniature pressure transducers located in a spanwise and chordwise array to provide measurements of the aerodynamic loading. Successes with this technique have included the RAE 9615 section (Lynx rotor blade section, Ref 13) and, more recently, the special modification to the tip planform made to the research Puma (Ref 14). These and many other studies outside the UK have spawned a formidable database on rotor aerodynamic characteristics and generated knowledge enough to be the sole subject of a lecture in the current series. Finally, test techniques for assessing performance degradation in adverse meteorological conditions are omitted. A review of the UK approach to testing in natural snow and ice can be found in Ref 15. Recent certification trials with the Chinook HC-Mk1 helicopter aimed at optimizing a heated rotor blade de-ice system are described in detail in Refs 16 and 17.

Three prevailing aspects deserve some preliminary discussion. These are the test atmospheric conditions, instrumentation requirements and data analysis and presentation. The international standard atmosphere (ISA) has been established to enable meaningful performance comparison to be made. Knowing the test height (pressure altitude) and outside air temperature (OAT), the density (σ), pressure (δ) and temperature (θ) ratios can be easily computed from simple formulae (Ref 18) or extracted from charts (Fig 1).

Needless to say, test atmospheric conditions need to be known fairly accurately, including the magnitude and direction of any winds (for hover tests) and turbulence levels. Apart from the measurement of altitude and OAT, the instrumentation required will depend on the test being performed. A bare minimum for the quasi-steady testing would be an airspeed indicator (ASI), engine torque metre and rotorspeed gauge. In addition, for low speed and hover testing, an accurate low airspeed indication is required unless ground reference testing is being performed, when additional ground equipment is needed. It is usual for aircraft that are to undergo a thorough test programme to have comprehensive instrumentation and to be fitted with onboard data recorders. Some capacity for an air to ground telemetry recording link is necessary for flight close to critical conditions, where on-line monitoring and recording of critical stresses is necessary.

All instruments need to be calibrated regularly in the varying test environments. For example, most pitot static tube arrangements for measuring the air dynamic pressure and hence flight speed, have calibrations that vary considerably with sideslip and

Similar tests are performed to establish the manoeuvre margin for load factors less than one using the push-over technique. For steady turn tests, the aircraft is again trimmed in level flight at the test airspeed. Load factor is applied incrementally by increasing bank angle at constant collective and airspeed, and maintaining balance with pedals. Cyclic is re-trimmed at each test condition and the tests conducted for both left and right turns. Rotorspeed should only be adjusted to remain within power-on limits and since high rates of descent may be achieved, care should be taken to remain within a defined altitude band (*eg* ±1000 ft of test conditions).

Fig 36 illustrates results that may be derived from these tests; the manoeuvre stability is deliberately shown to be negative at the higher speed. The cyclic to trim variation with load factor for the steady turn will typically be steeper than the corresponding pull-up result on account of the increased pitch rate in a turn for a given load factor. The relationship between cyclic to trim and pitch rate or load factor can be derived from the usual linearised theory in the form, neglecting flight path angle effects,

pull-ups:

$$\theta_{1s} = -\left(\frac{M_q Z_w - M_w V}{Z_w M_{\theta 1s} - M_w Z_{\theta 1s}}\right) q \qquad (9)$$

$$q = \frac{g}{V}(n - 1) \qquad (10)$$

turns:

$$\theta_{1s} = -\left(\frac{M_q Z_w - M_w V \left(\frac{n}{n+1}\right)}{Z_w M_{\theta 1s} - M_w Z_{\theta 1s}}\right) q \qquad (11)$$

$$q = \frac{g}{V}\left(n - \frac{1}{n}\right) \qquad (12)$$

Here θ_{1s} is the applied cyclic pitch (positive aft), q the pitch rate, V the flight speed and n the load factor. The derivatives will themselves vary with rotor thrust and rotor disc incidence and a more exact analysis is required for higher values of n. Nevertheless, equations (9) and (11) are valid representations of manoeuvre stability parameters. The numerator in equation (9) is the classical manoeuvre margin parameter that should be positive for 'stability' and acceptable handling characteristics. Typically an increasingly positive M_w variation with speed will lead to a deterioration in manoeuvre stability to the point where the margin can change sign. The load factor parameter in the manoeuvre margin parameter for steady turns arises from the inclination of the weight component from the fuselage normal and, clearly, at low bank angles will serve to reduce any undesirable effects of a positive M_w. At higher bank angles however any unstable tendencies will re-emerge.

The tests described above to establish the manoeuvre margin are carried out at constant collective pitch settings. In many practical situations however the pilot will use collective in conjunction with cyclic to maintain height. The pitching moment generated by collective application will be nose up and hence the cyclic position to trim will be further forward than indicated by the tests at constant collective unless control interlinks have been built in. This effect can be compounded by an increased download on the tail from the main rotor downwash. On other occasions, the pilot may choose to decelerate the aircraft in the turn, hence requiring increased aft cyclic displacement. This variability of stick position with load factor depending on the type of manoeuvre flown does not provide the pilot with a reliable tactile cue in manoeuvres. In any case, stick force per g is of more concern to the pilot, particularly in the mid-high speed band, and several recent designs (AH-64, Ref 43; SH-60, Ref 44) have force feel systems that provide a positive and reliable cue to the pilot of manoeuvre margin.

3.2.3 Lateral/directional static stability

The ease with which a pilot can coordinate entry to a turn, maintain trim in asymmetric flight or point the fuselage away from the direction of flight depend critically upon the ratio of two static stability effects, the yawing moment (N_v) and rolling moment (L_v) due to sideslip, or directional and dihedral stability respectively.

Estimates of these effects can be derived from steady heading sideslip flight tests at a range of forward speeds and normalised weight conditions from climbing through to autorotative flight. Such tests will also highlight any control problems within the sideslip envelope which is usually defined from fuselage stress considerations as a piecewise linear function of airspeed. Any pressure error corrections in sideslip need to be calibrated prior to the tests if a swivelling pressure head is not available, and the sideslip tests subsequently performed at defined EAS conditions. At each test point, control angles to trim and aircraft attitudes are recorded. Fig 37 illustrates a typical set of trim control results for varying airspeeds, with the slopes of the curves indicating directional and dihedral stability. Neglecting the rolling moment due to tailrotor thrust change, which in many cases is clearly not valid, the following ratios can be derived from the steady moment balance,

determined at a given speed by noting the new trim stick position for speed increments at constant collective pitch. The two results are shown in Fig 35, where for illustration, the true speed stability is shown to be negative and contrary to the apparent speed stability at the lower speed.

The test technique to investigate true speed stability is fairly straightforward. Having established trimmed flight at a defined airspeed and power setting the aircraft is re-trimmed in a series of speed increments below and above the test airspeed with cyclic. Alternation between positive and negative increments allows the aircraft to remain within a sensible altitude band (*eg* ±1000 ft) for level flight airspeed tests. For climb and descent conditions two passes through the required altitude band are typically required. While conducting these tests the pilot will also be concerned with related 'ease of trimming' issues, *eg* controller breakout forces, force gradients, etc. Particular attention should be paid to identifying strong non-linearities, for example discontinuities, in the speed stability and not to confuse these with adverse controller force characteristics or atmospheric disturbances.

Most certification requirements allow a limited degree of speed instability at low speeds, on the basis that the effect is not so critical here with the pilot normally controlling both speed and flight path with a combined cyclic/collective control strategy. At higher speeds, particularly for cold weather operation, adverse speed stability can limit the safe V_{ne} and careful testing is required to highlight any advancing blade Mach number effects. One such problem arises when a forward speed increment results in the centre of pressure moving further aft on the advancing blade outboard sections. This compressibility effect serves to twist the blade cyclically to give a nose down pitching moment on the aircraft which needs to be counteracted with aft cyclic.

Within the framework of linearised stability theory the speed stability of a helicopter is determined by the value of the effective derivative,

$$M_u^* = M_u - \frac{Z_u}{Z_w} M_w \qquad (8)$$

obtained from equations for the initial pitching moment due to a speed disturbance or the final steady state cyclic increment. The effect is usually dominated by the pitching moment derivative M_u which has a stabilising contribution from the main rotor. The fuselage and tailplane contributions will depend upon the trimmed incidence of these components. Tailplane effects can dominate in some situations. Ref 41 describes the adverse effect on speed stability caused by tailplane stall in climbing flight in the SA 365N helicopter. Fitting small trailing edge strips on the tailplane attenuated this effect but to guarantee speed stability for steep climbs in the range 80-100 kn an additional speed hold function was incorporated into the autopilot. Similar small design modifications to the tailplane leading and trailing edges were required to achieve speed stability for the BK 117 helicopter (Ref 38).

In addition to the speed stability testing described above, further tests are required to explore the cyclic trim changes with power settings at different speeds from autorotation to max power climb. These tests are required largely to check that adequate control margins are available in these conditions but will also highlight the essential features of flight-path stability. Although there are no general requirements concerned with helicopter flight-path stability *per se*, for aircraft roles that demand precise flightpath control, *eg* guided approaches, testing will need to be carried out to establish pilot control strategy for the various flight phases. Such tests are likely to be carried out in conjunction with the development of the associated displays. Collective is, of course, the natural control to counteract flight path errors, but above the minimum power speed the use of cyclic can achieve a similar effect. If the aircraft has, for example, fallen below the glidepath and is flying too fast, pulling back on the stick will eventually cancel both errors. Problems arise below minimum power speed where, although the initial effect of pulling back on the stick is to climb the aircraft, the new equilibrium state will be an increased rate of descent. Although normal control strategy should preclude such problems under 'controlled' approach conditions, for unguided steep approaches or emergency situations the pilot needs to be aware of the potential problems. At very steep descent angles the problem can be exacerbated by power settling effects (Ref 42) and ultimately the vortex ring condition (see section 2.3.3), where static stability characteristics are overshadowed by dynamic effects.

While speed and flight-path stability are concerned essentially with cyclic to trim requirements in 1g flight, manoeuvre stability is related to cyclic changes required in manoeuvres involving a change in normal acceleration, or the stick displacement (or force) per g. All handling requirements specify that this should be positive *ie* aft stick required to hold an increased load factor, and as a consequence, there should be no tendency to 'dig in' during turning flight. The manoeuvre stability can be determined in flight from either symmetric pull-up and push-over manoeuvres or steady turns and should be measured across the full range of operational conditions, *ie* speeds, atmospheric conditions, aircraft loading. Care should obviously be taken to avoid excursions beyond the manoeuvre envelope described in section 2.2.4. For the pull-up tests, the aircraft is trimmed in level flight at the test airspeed. With collective fixed, the aircraft is then decelerated with cyclic and then dived to accelerate back to the test airspeed. As the test speed is approached an aft cyclic step is applied to achieve the desired load factor and airspeed as the aircraft passes through a level attitude. The test is repeated with increasing increments of aft cyclic until the maximum permitted load factor is achieved.

3.2 Quasi steady flying qualities testing

3.2.1 Controller characteristics and control margins

A qualitative evaluation of a helicopter's flying qualities for a particular role will include an assessment of the mechanical characteristics of the pilot's controls. Such characteristics are normally defined in the requirements specification and encompass trim control, cyclic self centering, breakout forces and force gradients, deadbands and an evaluation of any adverse transient effects and control force discontinuities. Although much of the data on controller characteristics can be obtained from measurements 'on the ground', a flight evaluation is necessary and can normally be made in conjunction with other flying qualities tests. Breakout forces that are too high for example can inhibit the pilot from making small precise changes in flight path, and sluggish hydraulic systems can impede manoeuvrability. Above all the pilot will need to be assured that his controls are correctly harmonised for the various flying tasks.

Another major concern will be the ability to trim the aircraft, with adequate control margins remaining for manoeuvring, throughout the operational flight envelope. The interpretation of 'adequate control margin' depends upon the certification authority or Government test agency. Mil Spec 8501A, for example, demands that at the flight envelope boundary a longitudinal and lateral cyclic control margin enough to produce at least 10% of the maximum attainable hover pitch or roll moment should be available (Ref 4). Compliance with such a requirement is therefore reasonably straightforward to establish from flight test. The FAA adopt a more flexible approach on the basis that configurations have been tested where a 5% margin was sufficient and others where a 20% margin was inadequate (Ref 7). For FAA certification what is required from flight tests is a demonstration that, at the 'never exceed airspeed', V_{ne}, a longitudinal control margin sufficient to produce a 'clearly positive nose down pitching' is available. similarly a lateral cyclic margin to 'allow at least 30° banked turns at reasonable roll rates' must be demonstrated at the envelope boundary. Once again the flight test requirements are fairly straightforward although, regardless of the interpretation of adequate control margin, due care should be taken to avoid excursions beyond aircraft limits.

At the other end of the helicopter's flight régime, at low speed and in the hover, flight tests need to be carried out to establish out of wind hover and sideways flight trim control margins. Limitations here will usually relate principally to tail rotor control (power or thrust), although cyclic trim changes can also be marked. Once again, trim requirements and control margins that need to be demonstrated depend on the intended role of the aircraft. Minimum civil requirements (17 kn wind) are less severe than military requirements (35 kn wind) for obvious reasons, although clearly, the operational envelope can be expanded up to the condition where safe flight is possible. Testing needs to be carried out in light winds and with a pace vehicle to establish true airspeed in sideways flight tests. Fig 32 shows results for the BK 117 helicopter in sideways flight (Ref 38) demonstrating a capability far in excess of the 25 kn sidewind design objective. The trends in these results are typical for 'anti-clockwise' rotor helicopters, illustrating the high power requirements in right sideways flight and steep control gradient in left sideways flight. Extreme conditions that can be encountered in these two régimes are tail rotor blade stall in right sideways flight and the vortex ring state in left sideways flight. Flight tests need to be performed at the most critical loading, density altitude, rotor rpm, control rigging and wind direction to establish operating limitations. For most configurations it has been found that, as a result of ground effect and main rotor wake/tail rotor interaction, the critical azimuth position corresponds to right and left quartering flight (see Fig 33 from Ref 39). Ref 40 describes the flight test development and design modifications required for the AH-64 to optimise the empennage design to achieve a 45 kn trim, with wind from the critical azimuth. This frank account of an extended development programme highlights the importance of flight testing in areas where a significant performance improvement is being sought, while theoretical predictive ability is somewhat inadequate.

3.2.2 Longitudinal static and manoeuvre stability

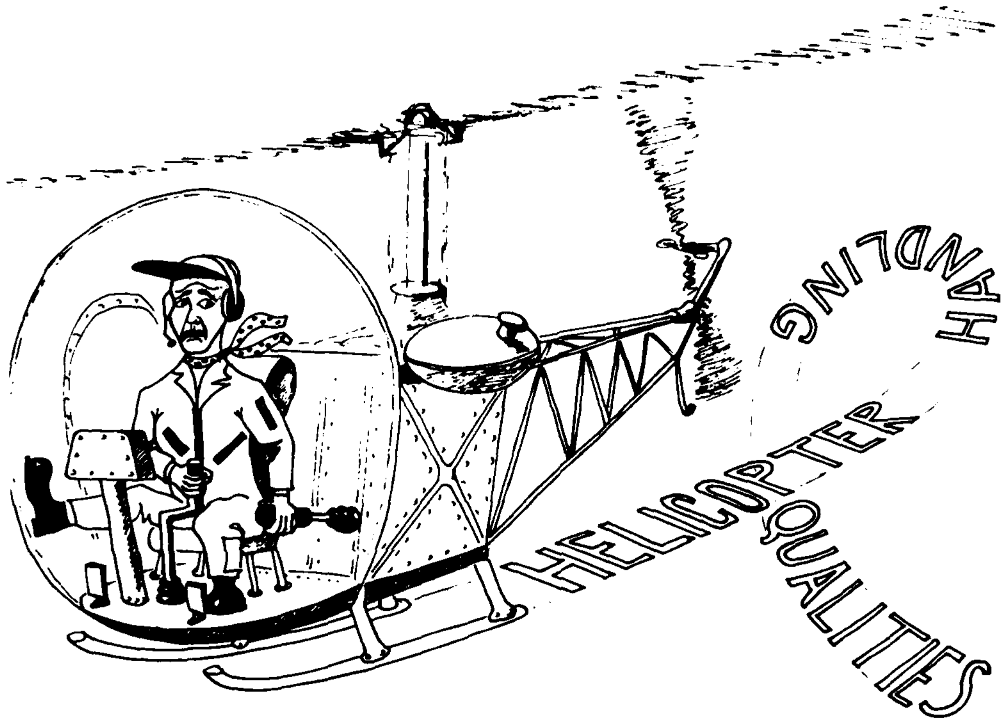
A pilot flying under IFR in turbulent conditions will have his workload significantly increased if, in attempting to control speed errors with cyclic, the new trimmed stick position is in the opposite sense to that initially required to cancel the perturbation. Likewise, when manoeuvring to avoid obstacles, a pilot will need to work harder if, having rolled into a turn and pulled back on cyclic to increase turn rate, he finds that he needs to push forward to re-trim in the turn. Both of these handling deficiencies of quasi-steady origin, are, generally speaking, unacceptable by any military or civil requirements standards and flight tests need to be performed to establish if they are present within the operational flight envelope. They represent negative margins of speed and manoeuvre stability respectively, that, together with their close companion flight path stability, form the topic of this section.

Fig 34 illustrates the consequences of positive and negative speed stability for cyclic control; in both cases the speed excursion is the same, but with negative speed stability the cyclic re-trims the 'wrong way'. There are two concepts traditionally associated with this characteristic, namely, apparent and true speed stability. The apparent speed stability is determined by the slope of the longitudinal cyclic trim variation with speed, *ie* with collective varying to maintain level flight or a defined rate of climb or descent. True speed stability, usually of more concern to the pilot, is

3 FLYING QUALITIES TESTING

3.1 Introduction

While performance testing is concerned largely with establishing operational efficiency, flying qualities testing is concerned with operational safety and the overriding condition of acceptance is that the operational pilot should be able to transit safely from one flight state to another, throughout the flight envelope, without excessive demands upon piloting skill and concentration. Operational requirements and compliance regulations need to be more quantitative, however, but in all their detail, they reflect and aspire to this one chief aim. Ironically, in testing for safety margins, the aircraft and crew are exposed to the great .t risk as unusual attitudes develop during stability tests, when recovery techniques are being learnt and critical control areas are being explored. Throughout this Chapter, in line with the philosophy of all test agencies, the 'incremental approach' to testing is emphasised, simply to minimise the risk and potentially hazardous consequences of 'cliff edge' type handling deficiencies.



The basic techniques of clinical testing for quasi-steady and dynamic stability and control are covered in sections 3.2 and 3.3. They include a treatment of how to convert the considerable database gathered during stability and control tests into engineering information; the methods of system identification are reviewed in this context and their role during development test flying and certification is considered.

It has long been recognised that, particularly for military helicopters, tailored handling characteristics serve to improve mission effectiveness. With the limited scope for major improvements possible through airframe design itself, such tailoring is largely achieved through automatic flight control systems and the related testing requirements are briefly considered in section 3.3. On the same theme of mission effectiveness, section 3.3 also deals with role-related testing and the interaction of flying qualities with pilot workload and task performance.

Comprehensive on-board instrumentation is normally required for flying qualities testing and, in addition, the test pilot's subjective impression provides an essential contribution to the test results. This is particularly true for role-related testing where thorough de-brief sessions are necessary to glean the relevant information from the pilot. Pilot rating scales and questionnaires are very useful in this context and provide results in a form that enable comparison to be made more easily. These should always be regarded as a summary of a detailed pilot report however, where a more complete account of flying qualities is collected. These issues are further discussed in section 3.3.6.

2.3.3 Vortex ring condition

At very low speeds and moderate rates of descent, depending largely on the rotor disc loading, a helicopter can enter a potentially hazardous flight state where high rates of descent can build up rapidly and erratic pitch and roll oscillations can develop. In addition, control effectiveness can change markedly, particularly collective control, with normal control recovery techniques seeming only to exacerbate the situation. Analogous to the stall in fixed wing aircraft in terms of the consequences to the flight path trajectory, but quite dissimilar in aerodynamic origin, the vortex ring condition is definitely a state to avoid, especially at low altitude and flight testing is usually carried out on a new configuration to determine the extent of the region and the associated recovery techniques.

The phenomenon has its origin in the peculiar flow characteristics that develop through the rotor in the intermediate range between the helicopter and windmill working states (Fig 30, from Ref 34). At very low flight speeds (< 15 kn) and rates of descent between 500-1500 ft/m, depending on the disc loading, the flow becomes entrained in a doughnut shaped vortex ring that leads to extensive recirculation in the outer regions of the rotor disc. The vortex ring is very sensitive to slight changes in flow direction and rapid fluctuating asymmetric development of the ring can lead to fierce moments being applied to the fuselage.

The standard recovery technique involves lowering the nose of the aircraft until sufficient speed is gained that the vortex is 'washed' away, then applying collective pitch to cancel the rate of descent. Different aircraft types all have their own peculiar characteristics in the vortex ring state. Early tests conducted at the RAE (Refs 35 and 36) produced results which varied from loss of control to mild wallowing instability. In general the aircrew manual for a type will contain an entry describing any particular features and advising the best recovery procedures. One such manual notes that rates of descent can build up to 6000 ft/m if vortex ring becomes fully established and that 'the aircraft pitches sharply nose down if rearward flight is attained'. Another refers to 'an uncontrollable yaw in either direction' eventually occurring. This manual adds that 'any increase in collective pitch during established vortex ring state creates a marked pitching moment and should be avoided'. All such references make it clear that considerable height will be lost if the vortex ring state is allowed to develop fully before recovery action is taken.

Interest in the effectiveness of collective control during recovery prompted a series of trials being carried out more recently at RAE Bedford using a Wessex 2 and Puma helicopters. The tests were qualitative in nature and aimed at exploring the behaviour of these two aircraft in the vortex ring state and establishing the benefits to recovery profile of increasing collective pitch before the aircraft nose is lowered to gain airspeed. The test technique options for approaching the vortex ring condition were somewhat constrained by the need to operate well above the ground (minimum height for initiating recovery action, 3000 ft agl) and the lack of reliable low airspeed measurement. The procedure adopted involved a deceleration from 50 kn to the hover, maintaining a constant pre-established (hover) attitude and rate of descent. The rate of descent was then increased incrementally until the vortex region was encountered (Fig 31). For both test aircraft the vortex region was quite difficult to find and apparently limited to a range of very low airspeed. With the Wessex the region was first encountered with the entry profile at 800 ft/m rate of descent. To quote from the pilot's report (Ref 37) "...with the rate of descent at about 800 ft/m we settled into the vortex ring; the rate of descent increased through 2000 ft/m in spite of increasing power to 3000 ft lb (hover torque reading). The vibration level was marked and a considerable amount of control activity was required to hold the attitude, though the cyclic controls always responded normally. Applying full power produced a rapid reduction of the rate of descent as soon as the rotor moved into clear air".

A major result of the tests was that applying collective prior to lowering the nose resulted in a height loss of about 150 ft during recovery whereas if the collective was lowered first and then increased when airspeed developed, the height loss was about 500 ft. Similar results were found with the Puma, except that the pitching and rolling moments were of higher amplitude and frequency and became more intense as the collective lever was raised during recovery. It is emphasised here that the results discussed above are particular to type and the beneficial use of collective during recovery may not read across to other aircraft. The difference in height loss during recovery for the two techniques is, however, quite marked and is operationally significant, particularly for low level sorties. Vortex ring is a real hazard area and can be encountered in a variety of situations, some less obvious than near vertical descents into restricted landing areas. If a pilot misjudges the wind direction and inadvertantly turns and descends downwind into a landing area, concentrating perhaps more on ground speed than airspeed, then he may fly dangerously close to the vortex ring condition. The final stages of a quick stop manoeuvre can also take the rotor through the vortex ring condition as the pilot pulls in power. Such manoeuvres are typically carried out close to the ground and the consequences of a delayed or inappropriate recovery procedure could be serious.

notional radius of a correctly banked turn corresponding to the maximum bank angle achieved, to define the Turn Agility Factor (TAF) as shown in Fig 25. Alternatively the TAF could be defined as the ratio of turn rate based on R_E to the maximum sustainable turn rate at the given speed. Lower TAF values will then to the higher thrust capability in transient manoeuvres. Tests were carried out with Puma and Lynx helicopters flying over a marked track on the airfield at Bedford. Entry speeds and aircraft weight were varied during the experiments and the aircraft position tracked using kinetheodolites, enabling track and height error to be determined. This simple test gives a direct indication of the maximum achievable thrust through R_E and a measure the effects of entry and exit rolling transients through the TAF. Clearly, a high value of TAF (approaching unity) and low value of R_E are desirable for agility. A summary of the R_E results for both aircraft is shown in Fig 26; trend lines are drawn through the data to indicate the sensibly linear variation of R_E with speed. Values of between 0.5 and 0.7 for the TAF were recorded for the Puma and, surprisingly, the values varied little with speed. It is suspected that the increased bank angles used at the higher speeds were a direct result of the increased transient thrust capability of the rotor. Unfortunately, the Lynx was not instrumented hence accurate estimates of bank angle were not obtained. However, the much crisper roll response in this aircraft conferred by the hingeless rotor would be expected to result in significantly reduced rolling transients, taking the TAF perhaps as high as 0.8. Estimated maximum bank angles were of the order of 75°. Times taken to complete the manoeuvre were again roughly constant with speed, and averaged out at 6 s for the Puma and 5 s for the Lynx. These manoeuvre times, the TAF metric and R_E are considered as reference figures for comparison with other helicopters over a similar speed range.

Further results from these tests and a detailed description of the test techniques can be found in Ref 32. Included are notes on piloting technique, sideslip effects on turn rate, control margins, height excursions and the effects of autostabilisation on the results. One interesting feature of the Puma tests was the peak fatigue damage rates attributed to the oscillatory pitch link loads at the maximum load factors in the region of 2.5 g. These were recorded at 2.5 times the level for infinite life (endurance limit, see section 2.2.4) of related head components. This damage rate occurs for only a short time, but has been logged in the cumulative running total kept for this research aircraft in its operation beyond normal limits agreed by the design authority. Clearly an operational requirement to enable regular exploitation of this level of agility will impact strongly on rotor and control system design. In addition to performance considerations the right angle turn tests demonstrated the potential for considerable control improvement in the mid speed range between 50-100 kn.

In the turning test described above an increase in rotor disc incidence through longitudinal cyclic control provided the principal rotor thrust increment. At lower speeds and in the hover, collective control provides this function and, for precise and crisp manoeuvring, there appears to be considerable potential for improving transient performance capability. As shown in Fig 24 the maximum thrust capability in the hover is quite high, although in practice, of course, power limitations and wake contraction effects severely limit the achievable value. However, for some very common manoeuvres, *eg* the bob-up and sidestep, high transient thrust levels are desirable and if the rotor and rotorspeed governor system could be designed appropriately, a much larger range of collective pitch could be usable.

A second series of flight trials has been carried out using the research Puma at RAE Bedford to explore and derive agility metrics appropriate to the bob-up manoeuvre. The test technique adopted is illustrated in Fig 27. The pilot's task was to climb with maximum power from the low hover position and re-establish a hover when the markers were in sight beyond the gate. Bob-up heights between 25-80 ft were explored for a range of different aircraft weights. Detailed analysis of the measurements obtained from on-board recorders and kinetheodolites is still in progress but some pertinent results have already emerged. Fig 28 illustrates the variation in height with time for a range of bob-up heights. Also shown in Fig 28 is the height trace corresponding to a maximum power climb to maximum rate of climb (approximately 34 ft/s). Maximum rates of climb achieved during the 25 ft bob-ups are only about 14 ft/s.

This result is largely a function of the vertical damping of the rotor that gives an effective time constant of a few seconds. If an agility metric based on the ratio of achieved climb rate to maximum possible climb rate were proposed then, clearly, for bob-up heights below 25 ft, values considerably less than 0.5 would result. To increase this value for small height excursions would require a significantly increased collective range together with some control over powerplant output and the resulting rotorspeed decay. Fig 29 shows the variation with time of selected parameters during the 25 ft bob-up described above. The initial high normal acceleration is not sustained and the rotorspeed and shaft torque vary considerably. The minimum allowable rotor rpm is 240 and, clearly, there is plenty in hand during the initial phase of the bob-up. This limit is, however, reached during the settling phase at the top of the bob-up.

Safe exploitation of increased agility in the hover would require a high energy rotor (Ref 33) to minimise rotorspeed droop, together with improved automatic control to protect against power, transmission and rotorspeed limits being exceeded. Once again the requirement for agility is seen to appear as a major factor in aircraft design.

The bob-up and right angle turn are examples of simple manoeuvre elements for which agility requirements can be quantified and test techniques established to validate the metrics and eventually demonstrate compliance.

specialised testing required to check a helicopter's role suitability is offered by the derivation of helicopter-ship operational limits. For every new helicopter-ship combination, recovery techniques need to be established for a range of sea states and wind conditions and safe operational limits defined. A description of the NLR approach to helicopter-ship qualification can be found in Refs 28 and 29. Another example involves the testing required to clear a helicopter to operate with underslung loads. Both horizontal and vertical drag can have a significant effect on performance. In addition, handling qualities can deteriorate rapidly in certain flight conditions as illustrated by an example in chapter 3.3 of this lecture. The UK (A&AEE) approach to the clearance of helicopter - underslung load combinations is described in Ref 30.

Battlefield operations represent one of the most demanding roles for a helicopter and the testing required to guarantee compliance with mission requirements can be exhaustive. Simple, task orientated performance tests would be helpful in this process and this theme is developed below as the principal topic in this section under the heading of helicopter agility.

Helicopter agility

Agility is that special combination of vehicle performance and handling qualities that enables the pilot to change rapidly the aircraft position or flight path with precision and safety. To describe agility in any more detail one needs to quantify the concept. As far as the principal force generator, the main rotor, is concerned a given level of agility is defined by the amount of excess thrust available and the speed with which this can be redirected. That agility depends both on performance and handling there should be no doubt; the performance aspect deriving from the magnitude of the thrust change while the handling aspect concerned more with the speed and precision of the magnitude and direction change. In addition, agility is only effective if it can be exploited safely. Indeed, however manoeuvrable a helicopter may be, if handling deficiencies or the need to carefully monitor flight envelope limitations inhibit or impose a high workload on the pilot then agility is lacking. The concept has been coined with special reference to the combat helicopter performing the hazardous task of flying undetected at nap-of-the-earth heights through the battlefield. Many of the elemental manoeuvres required in this role are closely associated with agility. For example (Ref 31), the unmask/bob-up, sidestep unmask/re-mask, dash/quickstop, pull-up/push-over and slalom are all precision flying tasks requiring high levels of agility.

The need to quantify agility arises in two ways. The aircraft designer, on the one hand, needs to know what the implications of a requirement for agility are for his design. Clearly, high agility is a very imprecise design goal. On the other hand, the service operator needs to appreciate the implications of agility on survivability and combat potential. From both perspectives trade off studies will need to be assessed and measures of agility established.

Fig 24 shows, in simplified form, the variation of rotor thrust limits set by the fatigue endurance (full-line) for a hypothetical helicopter in manoeuvring flight. As described in section 2.2.4, the rotor stall boundary is raised in manoeuvres so that, at moderate speeds the maximum thrust can be approached without incurring too much fatigue damage. The best manoeuvring speed will be affected by important design parameters such as disc loading and Lock number. The point is that if a high level of manoeuvrability is paramount the designer can build this into his aircraft albeit at the expense of hover and cruise performance.

Similar requirements could be established for transient thrust capability in the hover and at high speed but, again, the designer has to have data to work with. In order to accommodate precision, speed of response and safety with a given high performance level the designer will need to consider the implication on the flight control system design, and so on.

Design requirements for agility will be based on operational requirements laid down by the procuring authority and these will be established on the basis of trade offs in mission effectiveness. One method of quantifying such requirements is through the use of agility metrics for the class of relevant manoeuvres. Such metrics could be based on the time taken to perform a manoeuvre but are more useful if they can be closely associated with the aircraft performance and handling qualities required to fly defined flight paths. Normalised metrics suggest themselves as the ratio of achieved manoeuvre performance to the corresponding achievable maximum, a concept that is used widely to gauge quasi-steady performance efficiency (*eg* hover figure of merit). Demonstration of compliance with requirements would be simplified if agility could be presented in this manner and this leads to the role of flight testing. Test techniques need to be developed to validate the utility of proposed agility metrics and a database built up from the performance of current operational and research aircraft. With this aim in mind a test programme has been initiated at RAE Bedford to explore a range of task orientated manoeuvres and the classification of agility metrics together with the interaction of helicopter agility with pilot workload and handling qualities. Techniques relating more to the latter two aspects will be discussed in section 3.3 but, as already emphasised, it is impossible to completely divorce handling from performance when assessing agility.

One of the most primitive task elements studied was the right angle bend (Fig 25) representing a severe transient evasive manoeuvre (Ref 32). An effective turn radius (R_E) can be defined as the mean of the entry and exit legs. This may then be normalised by the

point (CDP) defines the speed/altitude point before which the aircraft must be able to land safely following single engine failure and after which the take-off can be safely continued on the remaining power. Landed safely refers to coming to a stop within the defined rejected take-off (RTO) segment, while a continued take-off amounts to being able to reach the take-off safety speed (VTOSS) with a defined climb rate, 35 ft above the ground.

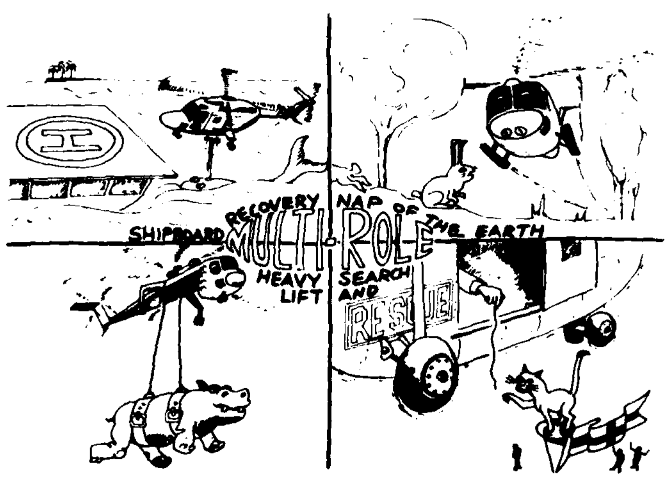
The test requirement and procedure are explained in detail in Ref 7, and are aimed at defining the CDP, evaluating piloting technique and determining the required take-off area at the various loading and atmospheric conditions for which the manufacturer is seeking certification. Similar diagrams and procedures define the requirements for Category A vertical take-offs and Category A landings. Clearly, an emphasis in all these tests is that the flight profile should be clear of the H-V boundaries, and experience has shown that the initial profile, immediately after the application of power, tends to be the most critical. Fig 22 shows test results for a BK 117 helicopter performing a Category A rejected take-off with number one engine failing at the CDP. The one second allowable delay time before collective reduction is shown in the figure and the RTO distance in this case was 900 ft (Ref 21).

As in the case of the H-V boundary tests, weight extrapolation based on simulation results is unacceptable to the FAA. Prediction methods are also improving as shown by several examples in Ref 21. Ref 24 also addresses the issue of predictive capability for a range of manoeuvring flying tasks using an energy model. The method is applicable in both design studies and certification flight testing. For the latter, regression analysis on the test data is suggested to determine unknown power coefficients defined in the energy model. Good correlation between test data and the prediction model is shown in Ref 24 for a number of cases. Fig 23 is typical, and shows the flight behaviour of an SA 330 Puma after failure of one engine during a Category A take-off from a platform. Although the rate of climb at VTOSS is slightly overestimated the predicted height at this condition is 23 m compared with the test result of 25 m.

The development of mathematical models, combined with increased usage of ground based piloted simulator facilities by manufacturers, will continue to increase the confidence in predictive ability and perhaps to increase allowable extrapolation of test data. For new designs, however, with new 'untested' features, certification agencies are likely to continue to demand extensive test flying in these critical areas for some time. Alongside any developments in this area must come progress towards standardisation of civil certification requirements internationally, though such progress should not inhibit different countries from being able to increase their own safety standards above a basic minimum. Rationalisation of flight test procedures and techniques would be one outcome of such a development, that would clearly benefit the helicopter industry (see Ref 27).

2.3.2 Role-related performance testing

The helicopter is a unique vehicle. The roles it is able to fulfil are expanding and its versatility will continue to be exploited by civil and military operators worldwide. At the same time operational requirements are becoming more demanding and the helicopter is nowadays far more than just a form of transport. It can embody a complex weapon system, form a very active element of an emergency rescue or law enforcement service, act as a mobile observation post or shipborne early warning system etc; the reality is that it can fulfill whatever the customer demands as evidenced by examples of the same type fulfilling a range of different roles. As a result of this versatility, performance data gathered from clinical testing, however essential for the operating data manual, will often not be sufficient for the customer, particularly the military operator, to gauge the utility of a candidate helicopter for a given role. This introduces a further dimension to flight testing that can be described as role-related or task oriented testing. As with so many of the dynamic test evaluations, aircraft performance per se will be only part of the story; handling qualities, crew workload and the ability to operate in adverse conditions will be equally important. A good example of the kind of



associated certification authority. In the coming years it will be the combined task of the helicopter manufacturer, operator and certification authorities to increase public awareness of the realities of helicopter transportation, the growing number of socially useful functions this type of aircraft is fulfilling and the improving safety record.

Flight testing plays three main roles in establishing reliable safety standards in this context; firstly through the determination of the required piloting techniques to enter autorotation or partial power flight following an engine failure, control of the aircraft during autorotative descent and the final flare and landing techniques; secondly, the definition of the height-velocity (H-V) boundary or avoid area, and thirdly, a demonstration that the helicopter is capable of carrying out the required category take-off and landing, safely. The flight tests required to cover these three areas generally involve some risk and need to be comprehensive in view of the limited extrapolation allowable by certification authorities.

Before the H-V boundary can be safely established for a new design, the test pilot needs to practice, at a safe height, procedures for entry into autorotation, flight idle glide or partial power flight together with landings from the same conditions. Initial tests will establish, for a range of initial conditions (airspeed, rotor rpm, altitude, AUW and cg range), the height change during entry and recovery phases, rotorspeed transients, effects on controls, changes in IAS, vibration effects etc (see Ref 23).

Secondary effects on aircraft systems will also need to be determined, *eg* any system failure at low rotorspeed. The effects of a delay in pilot reaction to engine failure can then be investigated for a similar range of initial conditions. The delay before lowering the collective lever should be increased incrementally until either the underswing in rotorspeed reaches the minimum allowable or a defined requirement is met *eg* UK Def Stan 970 (Ref 2) assumes a 2 second reaction time for the average pilot to identify a total power loss under unfavourable conditions, and reduce collective. Similar exploratory tests need to be carried out for both cases of total power failure and single engine failure in a multi engined aircraft, although for the latter the resulting rates of descent are sometimes significantly reduced. Piloting techniques for the other critical region of post failure flight, the flare and landing, can first be explored at a safe height, culminating in re-engagement, to 'bracket' the range of airspeeds, rotorspeeds and flare techniques best suited to the aircraft. Power-off or partial power landings can then be made for a range of different approach conditions, to cover a range of touchdown ground speeds from zero to the maximum allowed.

It is the responsibility of the manufacturer to determine the H-V avoid areas for a new helicopter. A thorough initial test programme, summarised above, will have been conducted to give the pilot confidence to approach the critical points on the H-V diagram (Fig 19).

The knee on the lowspeed boundary separates the take-off portion from the cruise portion. Below the knee normal pilot reaction time is assumed and performance is largely a function of rotorspeed decay rate and aircraft behaviour during run-on. The remaining lowspeed boundary above the knee and high speed boundary, are functions of pilot reaction time, hence height loss to achieve autorotation or partial power descent. As noted in Ref 23, the low speed curve is established with accelerating flight profiles into autorotation whereas the high speed boundary is established with quick stop type manoeuvres. It is hardly necessary to emphasise that as the boundaries are very real operational limits to safe flight, they must be approached with great care and established not on the basis of how well the skilled and practiced test pilot performs, but rather on how well a service pilot during a regular operation might be able to cope with the unexpected situation. Fig 20 illustrates the variation in critical parameters following single engine failure in a twin-engined helicopter from a low hover. Minimum touchdown rate of descent was achieved by allowing the rotorspeed to decay to the minimum allowable at the touchdown point.

The wide range of dynamic considerations, including piloting technique, that combine together to define the H-V limits for a particular helicopter make reliable predictions with simulation models difficult and this is reflected in the limited extrapolation allowable by certification agencies. For example, currently the FAA (Ref 7) do not allow any test weight extrapolation, and extrapolation based on an approved model to 2000 ft density altitude only is acceptable. Nevertheless the use of a predictive model is recommended to guide preliminary testing and properly validated tools can produce very encouraging comparisons with test data (Ref 21 and 24). The use of very simple parameters, like the 'autorotational index' (Ref 25), can also serve to establish the effects of gross weight, disc loading, density altitude and rotor inertia on autorotation and flare performance, during the design stage. In addition, new test techniques are being proposed (Ref 26) that may serve to simplify test procedures and reduce the risk associated with defining the H-V curve.

Although flight tests need to be performed to evaluate the effects of engine failure throughout the flight envelope, it is the take-off and landing phases, in the 'corridor' of the H-V diagram (Fig 19), where conditions are the most critical. Test procedures used to validate an aircraft's capability in this region depend upon the role the aircraft is intended for and hence on the certification authority. Civil requirements are the most demanding, typified by the Category A take-off and landing procedures for transport category helicopters defined by the FAA (Ref 7). Fig 21 illustrates the conditions that need to be met before certification can be granted. The critical decision

ultimate factors 1.125 and 1.5 respectively. Below this 'hard' limit are the 'softer' fatigue limits. Most modern service helicopters, particularly those operated in the battlefield tactical role, have a manoeuvre capability and operational requirement that demands fatigue loading in excess of rotor component infinite life. It is usual then, for the life of the critical components to be defined on the basis of a given percentage of an operational sortie spent beyond this endurance limit. Normally, if a specification is being met, the manufacturer will conduct the testing necessary to substantiate the fatigue life and hence loading spectrum required by the operator. Tests in turning manoeuvres are typical of those required to be flown and are usually initiated at speeds close to minimum power speeds with the pilot increasing the bank angle incrementally until a limiting condition is reached. This limit may be due to power, vibration, handling, too high a fatigue damage rate or even the limit load factor being reached. As in the case of the level flight tests, careful real time monitoring of the critical stresses is essential for this kind of test. Analysis of the stress data will enable the fatigue damage rates to be calculated and the critical components properly lifed. It is usual to find that fatigue loadings in turning flight are less than those obtained at corresponding thrust coefficients during level flight. This retreating blade stall relief is due to a redistribution of the lift on the rotor; the aft cyclic pitch required to generate the required aircraft pitching moment, increases the lift on the advancing blade with a corresponding decrease in incidence on the retreating blade.

Fig 18 shows a plot of thrust coefficient against advance ratio for a typical rotor derived from simple theory, which shows the loci of thrust for which the retreating blade has just stalled in the third quadrant. The stall relief in manoeuvring flight is clearly shown but the amount achievable in practice depends on details of the rotor design, particularly torsional behaviour. The limit load is also shown in the figure. At high speeds the rotor may have the capability to generate thrust in excess of the limit load, particularly in decelerating turns or in a pull-up from diving flight, and testing in these conditions requires care and caution.

Once the manufacturer has defined the safe manoeuvre envelope for a helicopter, then, if the vehicle is being procured for a specific military role, it will be the task of the Government test agency to define a, usually more restricted, operational flight envelope. Of course, at this stage, the scope of testing will encompass all that is required to demonstrate compliance with the defined service requirement. Close participation with the manufacturer during the development test phase will clearly minimise the amount of acceptance testing required but this is still likely to be extensive and several 'iterations' with the manufacturer may be necessary before the aircraft is fit for release into service. This general theme regarding the roles of the various test agencies has been covered in the Introduction to this lecture but is raised again here to distinguish between the manufacturer's safe or permitted flight envelope and the Service's operational flight envelope. In particular, it is the role of the Government test agency to recommend the limitations to be applied to Service operations, and, returning to the manoeuvre envelope topic, to recommend how the envelope limitations can be respected by the pilot. There are no universally accepted standards here and limitations found in aircrew manuals range from simple collective pitch limits to constraints on maximum permitted bank angles as a function of airspeed and from rotor mast bending moment constraints to cruise guide indicators that display critical stresses to the pilot. These and other varied solutions to the same problem have a common requirement that needs to be established during acceptance testing. They should give clear, unambiguous cues to the pilot that a critical condition is being approached. This is particularly true if the limiting parameters are not instrumented but rely on other pilot sensory cues, *eg* vibration levels or force feedback to the cockpit controls. It is widely recognised that the need to monitor aircraft limits, of whatever type, increases pilot workload, often during some critical evasive manoeuvre, and for future designs better solutions to the respect of limits need to be found. The benefits of a carefree manoeuvring capability (Ref 22) particularly for battlefield helicopters, would be extensive. Control systems that confer this ability need to be flight tested in a research phase before practical algorithms can be devised and the required level of confidence gained.

2.3 Dynamic performance testing

2.3.1 Take-off and landing performance and the height-velocity boundary

There is a misconception held by some of the general public that if a helicopter's engine fails, then a crash is inevitable. Such thinking naturally leads to a fear of flying in helicopters, however simple they make travelling, often fuelled by the accompanying myth that, in such situations, even the highly trained and skilled pilot can do little to alter the inevitable frightening course of events. To be fair, such public awe is usually balanced by the realisation that, particularly during brave and daring rescue missions, the helicopter is forced, where others have failed, to operate in very vulnerable situations and be exposed to many dangerous hazards that greatly increase the risk of accident. However mythical and misconceived these fears are, the safety record for helicopters, particularly during the first 25 operating years, gave some justification for public concern. Although accidents and incidents continue to occur as a result of engine failure, safety standards and records today are much better than before and will continue to improve, largely as a result of improved reliability and maintenance procedures, but also through a better understanding of the consequences to performance and handling of engine failure. This understanding is derived from flight test and the scope of testing required depends upon the intended role of the aircraft and the rules of the

2.2.4 Level flight airspeed and manoeuvre envelope

Helicopter airspeed and manoeuvre flight envelope limitations can be derived from a number of sources including power, transmission loads, vibration, rotorspeed, rotor stall and compressibility effects and control or handling deficiencies. It is the responsibility of the airframe manufacturer to define the envelope boundaries during development flight testing for certification purposes. The principal constraint on the airspeed/altitude/weight envelope of a helicopter is undoubtedly associated with the growth in fluctuating control system and rotor component loads as retreating blade stall is penetrated. As the rotorblade sweeps around into the retreating side of the disc, incidence is increasing to maintain the lift balance, (derived largely through cyclic pitch), and at the critical condition, lift stall is attended by a rearward shift in the centre of pressure of the stalled portion and hence a sharp and large change in the section pitching moment. The resulting abrupt change in blade torsional moment together with reduced and often negative torsional damping, can induce a stall flutter type of oscillation in the region of the retreating blade that decays rapidly as the blade penetrates the advancing region. When this condition exists, large fluctuating loads are generated in the blade control system. The correlation of blade stall with increased blade torsional response is illustrated in Fig 16 taken from Ref 20. The test was conducted on a Wessex helicopter at RAE Bedford with deliberately roughened blades to induce stall and simulate the effects of moderate leading edge erosion on unprotected light alloy blades. The direct correlation of trailing edge pressure divergence with lift stall and pitching moment 'break' is discussed in the reference. In Fig 16, the initial stall in the third (rotor disc) quadrant followed by a large amplitude oscillation in the pitch link load at the first torsional mode frequency is characteristic of the stall flutter phenomenon.

Operating the rotor at higher thrusts or airspeeds typically results in a rapid increase in the stalled region and in the peak to peak amplitude of the control system loads beyond the endurance limit corresponding to 'infinite' fatigue life. Although transient in terms of azimuth, they are repeated every rotor revolution and the effects of blade stall result in a very real rotor structural limitation, hence extensive flight testing is required to determine the flight envelope boundaries. Data extrapolation is least reliable for these critical rotor conditions and the combination of many detailed dynamic and aerodynamic interactions involving blade torsional response, blade/vortex interactions, unsteady and three dimensional aerodynamics also make prediction methods least reliable in this region.

Flight testing during the development test phase for rotor structural limitations requires extensive rotor instrumentation and an associated slip ring assembly. Monitoring of critical stresses during flight is essential for a safe approach to the critical condition and this is usually accomplished by test engineers at a ground telemetry station. The need for real time stress monitoring is particularly critical for modern designs that have irreversible powered controls and soft mounted crew seats, features that can rob the pilot of the natural cues that would give him some warning that critical conditions have been exceeded. Handling problems also accompany retreating blade stall but for single rotor helicopters, level flight performance is usually first limited by structural considerations as described above. An effective manoeuvre envelope usually means a degree of operation in conditions where fatigue damage is accumulated and components need to be lifed according to a defined spectrum of manoeuvres. Damage rate increases rapidly with load factor and handling deficiencies, particularly those which affect the control of normal acceleration, may well emerge as limitations to safe flight. Manoeuvring limits are discussed below and again in the flying qualities section of the paper.

The level flight airspeed envelope defined by structural fatigue limitations are presented in the operating data manual in the form shown in Fig 17, where the airspeed boundaries are shown as simple functions of density altitude and aircraft weight.

The boundaries are usually drawn at a defined percentage below the measured endurance limit to allow for flight in turbulence. Tests may also need to be performed to determine if the onset of advancing blade compressibility effects leading to pitching moment 'stall', occurs before retreating blade stall. The broken lines in Fig 17, representing the low temperature operating limits, are typical of such effects and indicate the rather abrupt nature of this aerodynamic limit.

In addition to defining level flight performance airspeed limits, the helicopter manufacturer must, before a design certificate can be awarded, flight test to define the safe manoeuvre envelope. These tests will normally be carried out with the same instrumented aircraft used for the airspeed tests and cover the full range of operating conditions. Strictly speaking, such testing will address much more than aircraft performance and should include such issues as handling limitations when manoeuvring at low speed as well as structural high speed limitations. In view of the intimate connection between handling and performance during manoeuvring flight a more complete discussion is continued in the sections on dynamic performance and flying qualities. Here we restrict the discussion to the topic of testing for structural limitations derived from rotor component fatigue and rotor and airframe strength considerations caused by 'normal' loads. Although extensive structural analysis and component rig tests will have been performed, a flight test programme must be undertaken by the manufacturer to cover the range of critical loading conditions and hence validate the integrity of the design. All helicopters have a defined limit manoeuvring load for structural design with proof and

the need to hold the test condition for a long period, perhaps up to two minutes if engine and fuel consumption data are required at the test point, it is useful to have any stabilisation or autopilot facilities switched on for the tests. A typical carpet plot derived from the test data points is shown in Fig 9.

A different method of establishing test points and covering the necessary range of rotor conditions, based on non-dimensional power and thrust coefficients, mean tip Mach number and advance ratio, has become an established test technique at the RAE Bedford. These quantities are more familiar in rotor performance theory and are easily related to the generalised parameters.

thrust coefficient $$t_c = T/\rho(\Omega R)^2 \pi R^2 s \quad (4)$$

power coefficient $$p_c = P/\rho(\Omega R)^3 \pi R^2 s \quad (5)$$

mean tip Mach number $$\bar{M}_t = \Omega R/a_o\sqrt{\theta} \quad (6)$$

advance ratio $$\mu = V/\Omega R \quad (7)$$

Here T is the rotor thrust, s the rotor solidity and a_o the reference speed of sound.

The test technique used involves defining a number of test conditions on a grid of $(t_c, \bar{M}_t)$, as shown in Fig 10, and establishing each condition for a range of airspeeds which can be converted later into advance ratios. Test charts, shown in Fig 11 for the RAE Bedford Research Puma, are required to adjust altitude and, if required, rotorspeed, during the testing to maintain constant t_c and $\bar{M}_t$. As shown on the chart, as fuel is burnt, then altitude needs to increase to maintain test conditions and as the temperature decreases, so rotorspeed should be decreased to maintain constant Mach number. Manual variation of rotorspeed can only be achieved below the rotor governor limit of 263 rev/min with the Puma, and this requires the observer/flight test engineer to operate the throttle levers to establish rotorspeed in concert with the pilot who establishes the test altitude and airspeed with collective and cyclic. Test points requiring rotorspeeds below about 250 rev/min are approached with a certain amount of caution as the effects of retreating blade stall can cause a rapid rotorspeed decay towards the lower limit of 240 rev/min. Testing in these limiting conditions also requires careful monitoring of critical rotor stresses via a telemetry link. Further aspects of the evaluation of flight envelope limits and associated test techniques are discussed in section 2.2.4.

A selection of results for the RAE Puma is shown in Figs 12 and 13, where the power coefficient is plotted against advance ratio for various values of t_c and $\bar{M}_t$. Trends can be seen, including the power rise due to compressibility at the higher Mach number in Fig 13.

Important ingredients of mission performance from the operator's viewpoint are the helicopter's range and endurance and the required testing needs to establish the best combination of speed and height to maximise range and endurance for a range of aircraft weights and atmospheric conditions. In addition, data needs to be gathered to determine the penalties incurred when the helicopter is operated in off-optimum conditions. The essential variable required to determine these efficiency measures is fuel consumption and this will normally be measured during level flight, climb, descent and hover performance tests. Results from the tests also allow payload/range charts to be produced for the aircraft operating data manual.

2.2.3 Forward flight climb and descent performance

The level flight power required curves in Fig 8 show clearly that excess power available and hence climb performance varies considerably with the forward speed of the helicopter. Testing needs to be carried out to establish the optimum climb schedule in order that a simple but efficient schedule can be recommended to the operator. The variation of the optimum conditions with AUW and ambient conditions also needs to be established. Similar test techniques can be used to establish the best airspeed for maximum range or minimum rate of descent in a flight idle glide or autorotation. A sawtooth test flight profile results in the most efficient use of test time with the first climb initiated at the lowest speed, followed by a descent, then a climb at a higher speed and so on until the highest required speed is reached. Repeating the test points at intermediate speeds and making the necessary measurements allows the results to be plotted in the form given in Fig 14. A mean curve can then be drawn shown in the figure to establish the mean AUW for the test points. Similar mean curves can be obtained at different test altitudes and an optimum climb schedule defined. Test results obtained during the descent portions of the profile can be used to derive a mean curve as shown in Fig 15, from which the airspeed for maximum endurance (minimum rate of descent(ROD)) or maximum range (best glide angle) can be derived, as shown.

Having established a climb schedule as described above it is usually necessary to validate the climb performance of the helicopter by conducting a ceiling climb test. Data on the performance at the absolute or service ceiling will also be required for the operating data manual. Such a test is often exposed to a host of other issues that affect performance as engine, transmission and control margins approach their limiting conditions.

ground referenced testing, particularly in high cold sites, care should be taken to adjust rotor speed, and hence the parameter ω, if blade tip Mach number effects are expected to be significant.

A typical carpet plot of 'smoothed' flight results from tethered hover tests is shown in Fig 4. As noted earlier, proof of compliance with a given requirement can usually be prescribed by Industry in the combined form of test data and theoretical predictions. Regarding hover performance prediction, theory is now developed to the point where very accurate predictions are possible across a wide range of altitude and temperature. The results in true non-dimensional form shown in Fig 5, were measured on a Wessex helicopter at RAE Bedford (Ref 20), and illustrate the increased rise in torque required at the higher thrust coefficients as the blade tips penetrate into drag rise conditions. For these tests a simple and effective method for establishing true hovers at altitude was practised whereby a ball on a string was tracked beneath the helicopter with the observer relaying the bow direction to the pilot.

Fig 6 is taken from Ref 21, and also demonstrates this prediction capability for the BK 117 helicopter. Similar results are presented for hover IGE. Also, in Ref 21 Mach number effects on power are shown to be predictable conservatively to within 10% at temperatures as low as -30°C. Greater acceptance of such predictions by clearance authorities will reduce the amount of time required to gather these somewhat imprecise test points. Currently, FAA regulations (Ref 7) stipulate that all OGE hover data must be obtained at the same test sites that the IGE data is obtained, and extrapolation is only possible up to 4000 ft.

The measurement of helicopter vertical climb performance can also be achieved using both ground referenced and air referenced techniques using ballasting and altitude respectively to effect changes in the normalised weight. The results will, however, usually exhibit considerably more scatter than the hover data for several reasons. The test points are difficult to set up accurately, wind speed and direction may vary with height and the rotor speed will tend to droop with a power increase thus affecting the normalised weight parameter. For ground reference testing the need for continued re-ballasting to achieve a constant weight parameter is clearly time consuming and a test technique that obviates this has therefore been developed. From a given reference hover OGE, rates of climb can be established for increments in power up to maximum torque. If these test points are then repeated in reverse order results of the form shown in Fig 7 can be obtained and a mean trend established. Vertical climb performance data will enable an operator to determine the power or thrust margin required, for example, to rise vertically out of restricted spaces, or to clear safely a moving surface, *eg* ship's deck. Steady state performance in this respect is not the only criteria that some operators use to establish the effectiveness of their helicopters in vertical manoeuvres. The initial stages of the bob up task, used by battlefield helicopter pilots to unmask from cover, involve transient acceleration and rotor speed effects that can dominate over the steady state characteristics. These and other issues relating to helicopter agility and associated handling qualities are dealt with in later sections.

2.2.2 Level flight performance, range and endurance

Level flight performance is determined by the power required throughout the speed range for the required conditions of all up weight (AUW), altitude and temperature, and rotor speed. Such performance data will appear in the aircraft's operating data manual in the form of a family of curves for each different atmospheric condition typified by Fig 8. Flight speed becomes an additional variable in the tests, in the form of the advance ratio μ ($V/\Omega R$) or simply (V/ω). Compressibility effects will become important, especially at higher forward speeds, and the normalised power coefficient can be expressed in functional form as either,

$$\frac{P}{\delta\sqrt{\theta}} = f\left(\frac{V}{\omega}, \frac{W}{\delta}, \frac{\omega}{\sqrt{\theta}}\right) \tag{2}$$

or

$$\frac{P}{\sigma\omega^3} = f\left(\frac{V}{\omega}, \frac{W}{\sigma\omega^2}, \frac{\omega}{\sqrt{\theta}}\right) \tag{3}$$

where V is the flight speed and δ is the atmospheric pressure ratio.

The normalised parameters in (2) are used if rotorspeed is variable and testing at constant pressure altitude preferred. Clearly a large number of test points will need to be flown to gather the data required and these may need to include points at various centre of gravity (cg) positions and with externally carried role equipment. Careful test planning is therefore required to economise on test time and charts will need to be prepared indicating to the aircrew the height to fly versus test AUW and rotorspeed versus OAT. The charts will normally be further broken down to define those test conditions likely to be available at the test location.

The optimum piloting technique used to stabilise on a test point will depend on whether the airspeed is above or below that required for minimum power. Above minimum power speed, collective is held fixed and rate of climb nulled and speed stabilised with cyclic. At and below minimum power speed it is easier for the pilot to first stabilise airspeed with the cyclic and thereafter null the rate of climb with collective. Due to

incidence. Position error correction is of paramount importance and a reliable method of overall calibration is still that of accurately timing over known distance in measured low wind conditions with sideslip and incidence as controlled variables. This procedure may change in the future with new techniques based on doppler laser technology. Fig 2 shows a typical pressure error correction. Because of the close relationship with flying qualities, dynamic performance testing usually requires a wider range of on-board measurements including pilot's controls, fuselage attitudes, rates and accelerations. In addition, ground based tracking equipment *eg* kinetheodolites, are mandatory for some tests, *eg* certifying for Catetory A take-offs.

Flight testing can generate considerable amounts of data and efficient forms of data reduction and analysis are required to convert this into useful information. Techniques vary from in-flight reduction of the test engineer's knee-pad data to extensive post-flight, or even real-time (Ref 19), computer analysis using a database management system together with a range of standard analysis packages. Both approaches are generally necessary, the former to guide the test programme in real-time and the latter to verify test points and produce definitive results. A desirable aim should always be that test data analysis is rapid enough not to impede but rather to guide the test programme and that all proceedural operations are automated.

2.2 Quasi-steady Performance testing

2.2.1 Hovering and vertical flight performance

For many helicopter operators good hover performance is a first priority although this will usually be compromised by rotor design requirements for high forward speeds. Often quite specific targets are required by an operator in terms of maximum all up weight that can be lifted, out of ground effect (OGE), at different atmospheric pressures and temperatures. These target points must then lie within the performance boundaries defined in the flight or operating data manual (ODM). Fig 3 shows a typical presentation of such data and the flight testing required to establish the operating boundaries is both exhaustive and varied. Accurate measurement of hover performance requires considerable pilot skill and test team coordination in view of the wide range of factors that influence the power required, and the difficulties involved in establishing a true hover. The factors include outside air temperature and humidity, altitude, aircraft condition and wind speed. The relationship between the power required and these factors can be expressed functionally in normalised form, dependent on thrust coefficient and tip Mach number, *ie* ,

$$\frac{P}{\sigma\omega^3} = f\left(\frac{W}{\sigma\omega^2}, \frac{\omega}{\sqrt{\theta}}\right) \qquad (1)$$

where P is the input power
W is the aircraft weight
σ is the atmospheric density ratio
ω is the ratio of actual to reference rotorspeed
θ is the atmospheric temperature ratio

For hover in ground effect (IGE) performance, an additional variable in this functional relationship will be hover height or height ratio (h/R) where R is the rotor radius.

To enable a wide range of normalised weights to be covered during testing a number of test techniques have become established. These fall naturally into two groups:

(a) ground referenced free flight and tethered hovering

(b) air referenced free flight hovering

A pre-requisite for the ground referenced techniques is that tests are conducted in very light winds (< 3 kn) since wind and turbulence and the attendant unsteady aircraft motion give rise to considerable scatter in the results. The free flight technique is relatively simple but requires a significant time spent 'on the ground' between test points changing the aircraft weight. The tethered hovering technique, whereby the helicopter is attached by cable to a suitable tethering point, (cable tension relating directly to rotor thrust), removes the need for much of this time consuming activity and simplifies height control. However, it becomes important in this type of test for the pilot to ensure that the cable is vertical and hence position control becomes more demanding. In addition, both ground referenced techniques normally require at least two test sites to cover the required range of normalised weight.

Air referenced test techniques have been developed to overcome the problems associated with weather and the need to test at different sites that attend ground reference testing. Early testing techniques involved formating on neutrally buoyant 'objects' such as balloons, smoke puffs or even clouds but with the advent of reliable low airspeed indicators, calibrated to work in the rotor downwash, these rather crude techniques have been largely superseded. Height control remains a problem however, and some care must be taken to avoid the vortex ring condition; climbing the last stage to the test condition is therefore favoured. A wide range of non-dimensional weights is best covered by flying the test points in a descending sequence from high altitude, high weight to low altitude, low weight. In addition, the test altitudes should be planned such that engine limits do not restrict the top end of the normalised weight range. As with the

$$\frac{\delta\eta_{1c}}{v} = -\frac{L_v}{L_{\eta 1c}} \tag{13}$$

$$\frac{\delta\eta_p}{v} = -\frac{N_v}{N_{\eta p}} \tag{14}$$

where $\delta\eta_{1c}$ and $\delta\eta_p$ are the pilot's control deflections from level trim, v the sideslip velocity and $L_{\eta 1c}$ and $N_{\eta p}$, the rolling and yawing moments from lateral cyclic and yaw pedal respectively. Assuming that these control derivatives vary little with speed, the trends, though not absolute variations, in dihedral and directional stability can therefore be derived. Fig 38 shows the variation of cyclic and pedal trim positions in sideslip for a climb, level and descent condition measured on the RAE research Puma at 100 kn. A strengthening dihedral effect and weakening directional stability are apparent in the climb conditions. These particular characteristics are referred to again in section 3.3 when considering the effect of flight condition on dutch roll damping. The directional stability for small sideslip angles tends to be very sensitive to flow conditions at the fin, in addition to the detailed profile of this surface. Ref 41 describes how major improvements to directional stability were achieved during test flight development of the AS 332, AS 355 and AS 365 helicopters through relatively minor design modifications to the fin section.

For helicopter configurations with a high set tail rotor the rolling moment from the tail rotor will contribute significantly to the lateral cyclic required in steady sideslipping flight. When the dihedral effect is small the trimmed cyclic may be in the same direction as the pedal trim. An overriding pilot consideration when testing for directional and dihedral stability should be that clear unambiguous sideforce cues indicate the direction of sideslip. The pilot needs to be clearly alerted when sideslip limits are approached. Test techniques for isolating the dihedral stability for helicopters with strong tail rotor rolling moments are described in Ref 10.

Just as speed stability can be determined by perturbing the aircraft with longitudinal cyclic in the vertical plane, so spiral stability, or the tendency of the aircraft to return to or diverge from level flight when disturbed in roll, can be derived from the 'turns on one control' test technique. Having established the desired 'wings level' trim condition, lateral cyclic is used to roll the aircraft to a small bank angle. Speed is held constant with longitudinal cyclic and the lateral cyclic re-trimmed to hold the new bank angle and turn rate; rudder and collective are held fixed. The manoeuvre is repeated in the opposite direction and for a range of steady bank angles. Similar tests can be performed using yaw pedal to initiate and trim in the turn. For both tests the control deflection required to maintain a steady turn gives a direct indication of the aircraft's spiral stability. If out of turn control is required then the aircraft exhibits spiral stability with the converse also being true. Once again, using linearised derivative theory and combining terms in the rolling and yawing moment equation in the steady turn, the control perturbations can be written,

$$\delta\eta_{1c} = \frac{(L_v N_r - L_r N_v)}{L_{\eta 1c} N_v}\, r \tag{15}$$

$$\delta\eta_p = \frac{(L_v N_r - L_r N_r)}{N_{\eta p} L_v}\, r \tag{16}$$

Here r is the yaw rate in the turn and the additional assumption is made that rolling moments generated by aircraft pitch rate are negligible. From the test results the ratio of the spiral stability parameter to the control derivatives can be derived as a function of flight conditions. For spiral stability this parameter must be positive implying control deflections into the turn.

3.3 Dynamic flying qualities testing

3.3.1 Dynamic stability

The natural dynamics of helicopter flight evolve from a complex interaction of aerodynamic and dynamic forces that vary markedly in their make-up throughout the flight envelope. In some situations, coupled oscillatory motions can dominate the response, whereas in others, sharply aperiodic divergences can lead to rapid and large changes in flight path and speed. Flight testing to establish natural stability characteristics are required to highlight potential handling problems or to help explain known handling deficiencies, to compare aircraft characteristics with requirements and to provide data for the design of stability augmentation systems. The tests described below cover both long and short period oscillatory modes as well as aperiodic modes and refer only to stick fixed characteristics. Requirement specifications commonly distinguish between longitudinal and lateral stability in this context and the discussion below follows this tradition.

An unstable low frequency oscillatory mode involving changes in aircraft pitch and speed characterises the long term stability of helicopter longitudinal motions. This mode can take the form of a mildly unstable pendulum-type motion in the hover to a rapidly divergent 'phugoid' type oscillation at high speed. Aircraft centre of gravity position

has a marked effect on the stability in this mode in forward flight, reflecting the contribution of rotor thrust tilt to aircraft pitching moment particularly in the presence of large fuselage and tailplane moments. At forward cg extremes the oscillation can actually stabilise at moderate speeds whereas, for some configurations, particularly hingeless rotor helicopters or helicopters with small tailplanes, the oscillatory mode can split into two aperiodic divergences at high speed. The mode differs from the traditional fixed-wing 'phugoid' in that the speed changes during the 'climbs and dives' induce pitching moments, that cause significant variations in fuselage and rotor incidence and thrust. In the early days of helicopter testing these differences were often a source of surprise to those whose background was essentially fixed wings. Ref 45 describes a loss of control incident on an S-51 helicopter at RAE. The pilot had excited the phugoid mode with a longitudinal cyclic pulse; recovery action was initiated at the end of the third oscillation, the aircraft increased speed in a dive and during the pull out the blades hit the droop stop, and eventually the fuselage, causing a rapid uncontrollable rolling motion. The resulting erratic motions, during which the pilot became disorientated, eventually settled down and the aircraft was flown back to RAE and landed safely. The auto-observer recorded a peak normal acceleration of more than 4 g during the manoeuvre causing severe buckling to occur in the rear fuselage. Two of the conclusions of Ref 45 were:

1...'that large rapid movements of the controls (particularly backward) are to be avoided at high speed', and

2 'Some form of flight testing techniques should be devised whereby the susceptibility of a helicopter to this trouble could be ascertained in the prototype stage'.

Today, the first conclusion is just as relevant as it was 40 years ago and test techniques are now established to highlight the important features of helicopter long term stability. Having established a trim condition a pulse type input in longitudinal cyclic is applied, returning the control to the trim position. The period of this mode is usually greater than 12 seconds, hence depending on the degree of instability, long response times are required to estimate the frequency and damping. Very smooth conditions are needed as inputs from turbulence contaminate classical motion during such long recording periods. The cyclic input amplitude and duration may need to be varied to produce a sufficient perturbation in flight path or airspeed to obtain a reasonable excitation of this mode. Because of strong nonlinear effects with airspeed and incidence, the modal characteristics can be expected to vary markedly with input size. Dynamic stability criteria vary between requirement specifications although most define boundaries of acceptability for VFR and IFR flight in terms of period and damping ratio.

Short term longitudinal dynamic stability characteristics dominate pilot impressions of handling qualities relating to pitch attitude control. In low speed and hovering flight the pitch and heave motions are essentially decoupled and short term stability in each axis is described by the principal damping derivatives M_q and Z_w. In forward flight these motions become strongly coupled and for most helicopters, adopting fixed wing parlance again, a short period pitch/heave mode emerges, approximated by the familiar response equation, in which speed is assumed constant.

$$\ddot{q} - (Z_w + M_q)\,\dot{q} + (Z_w M_q - M_w(Z_q + U_o))q = M_{\theta 1s}\,\dot{\theta}_{1s} - (Z_w M_{\theta 1s} - M_w Z_{\theta 1s})\theta_{1s} \qquad (17)$$

It should be noted that this form of approximation is not always valid and for some configurations with a positive (unstable) M_w and strong M_u, the short term stability characteristics are determined by a combination of stable aperiodic modes and an unstable long period oscillation. Ref 46 discusses the range of application of the short period approximation given by (17). Whatever the character of short term stability a convenient test control input which minimises excitation of the long period mode is the cyclic doublet. Conducting a cyclic frequency sweep initially will enable the pilot to establish if a clearly defined short period oscillation exists. This can be identified by the frequency at which aircraft response becomes out of phase with the control input, and of course, where the pitch response is a maximum. Using control jigs or fixtures, cyclic doublets of various amplitudes can then be applied at the modal frequency. In addition to the frequency, the damping ratio or number of oscillations to half amplitude characterises this mode. Although control response itself is discussed later under a separate heading it is usual to combine short period mode testing with an investigation of the ease or difficulty in making small precise pitch attitude changes. Asymmetric doublet inputs serve to initiate the pitch attitude change and the pilot should then determine what corrective action is required, to subdue overshoots for example. The frequency and magnitude of the cyclic inputs can be varied to achieve the best performance for this task. Fig 39 illustrates the pitch response of a Puma to a cyclic doublet at 100 kn. For aircraft that do exhibit a distinct short period oscillatory response the frequency increases with airspeed (term proportional to q in (17)) while the damping increases slightly as Z_w becomes more negative. Both these parameters vary with aircraft cg position, AUW and density altitude and tests are required across the range of these conditions.

In forward flight, lateral/directional dynamic stability is determined by the characteristics of the oscillatory 'dutch roll' mode and the aperiodic 'spiral' mode. As with fixed wing aircraft, the contribution of these modes to long and short term response

is critically dependent on the values of the two static stability parameters L_v and N_v (see also section 3.2.3). Fig 40 shows the relevant two-parameter stability diagram with the spiral and dutch roll stability boundaries drawn in, along with sketches of typical short term roll responses for various parameter combinations. Summarising the results, weak directional stability and strong dihedral can lead to dutch roll instability while the converse leads to spiral instability. A compromise is therefore necessary for acceptable all-round handling characteristics. It is often the case with helicopters that weak directional stability results from aerodynamically inefficient fins that have necessarily thick aerofoil sections and operate in the wake of the main rotor and upper fuselage. Dutch roll damping suffers as a consequence but, as described in section 3.2.3, a significant increase in N_v can be achieved through minor design modifications to the trailing edge of the fin that effectively decrease the trailing edge angle. Further improvements can be derived from increasing the fin area with endplates on the horizontal tailplane. Fig 41 from Ref 38 illustrates the marked improvement in dutch roll damping, that met the FAA single pilot IFR oscillatory mode requirement, through the fitting of endplates on the BK 117 helicopter. Various approximations have been proposed for the dutch roll frequency and damping depending on how much yaw, roll and sideslip are present in the mode. For pure yawing motion, damping and frequency can be simply derived from the yaw derivatives N_r and N_v. This model has limited application, however, in view of the normally strong dihedral effect L_v inducing considerable rolling motion. A model that includes roll/yaw coupling is discussed in Ref 47 and shown to give good agreement with flight results for the cases considered. The frequency and damping approximations take the form,

$$\omega^2 = U_o\left(N_v - L_v \frac{N_p}{L_p}\right) \tag{18}$$

$$\zeta = \frac{1}{2\omega}\left(- N_r + N_p\left(\frac{L_r}{L_p} - \frac{U_o L_v}{L_p^{\,2}}\right)\right) \tag{19}$$

In equation (19) the only way the directional stability parameter N_v can affect the damping ζ is through the effective dihedral L_v when the product of inertia I_{xz} is non-zero. For configurations where sideways motion or longitudinal couplings contribute significantly to the motion the above approximation will be invalid.

Optimum test inputs designed to excite the dutch roll mode clearly depend on the type of motions involved and vary from pedal or cyclic doublets or oscillatory inputs to cyclic or collective pulses. Lateral cyclic inputs, for example, will expose the degree of adverse or proverse yaw (N_p) present whereas collective inputs will isolate the dihedral effect on the motion, unless strong collective/roll coupling is present. Fig 42 shows the dutch roll response of the RAE Research Puma following pedal doublets in the three conditions shown; descent, level flight and climb. Ref 41 describes how modifications to the fin on the SA 332 Super Puma significantly improved the dutch roll stability characteristics of this aircraft compared with the original Puma design.

Dutch roll characteristics tend to be fairly sensitive to flight path angle as indicated by Fig 42. The effect of steady sideslip on damping can also be significant, arising from nonlinear interactional effects with the main rotor wake (Ref 47). Clinical dutch roll stability tests should be carried out in these various conditions. Analysis of the oscillatory characteristics, including phase angles between the various degrees of freedom should enable fundamental parameters associated with the motion, such as yaw damping and adverse/proverse yaw to be estimated.

The characteristics of the spiral mode will determine the tendency of aircraft to return to, or depart from, a level trim condition following a perturbation in roll. As noted earlier, spiral and dutch roll stability are naturally at variance with one another so that a strongly stable spiral mode will result in attitude demand type roll control and strong excitation of a weakly stable or unstable roll/sideslip oscillation during simple uncoordinated turns. Spiral stability can be determined qualitatively from the 'turns on one control' technique described in section 3.2.3. The test techniques recommended in Ref 7, involve establishing an out of balance trim condition, returning controls to the level trim condition positions and measuring the bank angle response. As pointed out in Ref 7, of particular interest is the time for bank angle to pass 20° and this time should not be so short as to cause the aircraft to have objectionable flight characteristics in the IFR environment. The time period to double amplitude (20°) should be at least 9 seconds. For aircraft with marginal spiral stability the range of flight conditions covered in the testing may be relatively large with left and right spiral stability explored in some detail.

Dynamic stability testing can be a hazardous task; it is preferable for serious handling deficiencies to be discovered in tests through the incremental approach rather than when the aircraft is undergoing a role-related test phase. This may, however, be difficult to plan for in a time-critical development phase, because of the extensive number of test hours required to cover the complete envelope. Perhaps of greatest concern are the stability variations during manoeuvring and asymmetric flight conditions, where simple augmentation systems can become saturated, and the consequences of an unexpected change in stability to a pitch or roll/yaw divergence can have a serious impact on flight path and structural loads.

3.3.2 Control response

Military helicopter handling requirements both in Europe and the US are widely felt to be out of date both in their structure and in many of the detailed specifications. The revisions currently being prepared are expected to be considerably more relevant to the wide range of operational roles fulfilled by helicopters and should address particularly the requirement for tailored handling characteristics for different flight phases and usable cue environments. Ref 31, for example, proposes detailed requirements for control response types that can be acceleration, rate, attitude or translation rate command in character. The adoption of a much more sophisticated requirement for control response characteristics will impact significantly on aircraft/control system design as well as flight test techniques. The short term response to control inputs define to a large extent the aircraft's manoeuvrability and clinical control response testing, perhaps more so than stability tests, gives the pilot a clearer impression of short term handling and how well the aircraft will perform during role-related tests.

The test input to evaluate control response about all axes throughout the flight envelope consists of a step, or more practically a ramp, applied from the trim position and held against a fixture or scale. For short term response evaluations recovery can be initiated when the steady state response in the required axis is obtained or when a pre-determined flight parameter limit is reached. A range of increasing input sizes, in both control directions, should be applied, to test for linearity and consistency. The type of response will vary with control type and flight condition but the essential features of interest will be:

- i control sensitivity - acceleration/unit control input,
- ii delay in build up to maximum acceleration,
- iii steady state primary response (rate (velocity) or attitude (position)),
- iv time constant of primary response - damping,
- v tendency to overshoot,
- vi cross coupled responses.

These features are illustrated in Fig 43 and although a detailed analysis of recorded data will be required to quantify these effects, immediate qualitative pilot impressions are useful to establish the adequacy, consistency and predictability of the response. Tests are normally carried out 'one control at a time', with the remaining controls fixed at their trim values, but these can be supplemented with tests where the pilot attempts to minimise coupling excursions with secondary controls. The effect of this action on the primary response characteristic may be important and should be noted.

Control response in hovering flight is typified by a first order lag type response in all axes and dominated by the principal damping parameters L_p , M_q , N_r and Z_w . One exception to this rule is the heave response in ground effect where an oscillatory second order response is caused by the ground cushion. Short term angular motion response is also modified after a few seconds as translation rate builds up and pitch or roll moments develop to reduce the response. In forward flight roll control response remains dominated by the roll damping L_p and, of course, control sensitivity, that remain fairly constant with airspeed. As noted in earlier sections, however, the response can soon become oscillatory if roll motion excites the dutch roll mode leading to sideslip excursions. Forward flight has a stronger effect on longitudinal control response (cyclic and collective) as the short period mode develops and both pitch and heave excursions take place. Also, for larger cyclic inputs, nonlinear speed effects can modify the control response and reduce the achievable steady state pitch rates. Figs 44 and 45 illustrate the cyclic control response for the RAE Research Puma at 100 kn. Following a step input in longitudinal cyclic (Fig 44) the pitch rate is seen to achieve a maximum value in about 2 seconds while the peak normal acceleration is delayed somewhat as the incidence continues to increase. Roll coupling is seen to be strong, arising mainly from the pitch rate (L_q) effect. Roll rate response to a lateral cyclic step (Fig 45) rises more rapidly as expected ($|L_p| \gg |M_q|$) but soon subsides as sideslip builds up and the oscillatory mode begins to respond.

3.3.3 Gust response

For helicopters without any form of control augmentation the features that help make the response insensitive to atmospheric disturbances, *ie* strong stability, serve also to impede manoeuvrability. The compromises between stability and control are partially resolved in many current designs through a combination of stability and control augmentation that essentially gives the aircraft different response characteristics to pilot commands and external disturbances. This facility gives the designer greater freedom to tailor handling qualities, but to fully exploit this freedom good test data on gust response characteristics are required. During gust response testing the pilot will be interested in the types of disturbance (*eg* moderate turbulence, including discrete gusts and windshear) from which he can regain control, or maintain control long enough to complete his task or fly out of the disturbance. He will be interested in which dynamic

modes are excited and whether the response cues are adequate to be able to judge the required control strategy for recovery, and he will need to estimate the increase in workload required for continuing precision flying tasks in turbulent conditions. An example might be the effect of gusty conditions on handling criteria established for smooth conditions, *eg* precision hover task. Finally, the pilot will be required to judge the ride quality of the aircraft in gusty conditions and quantify the degree of tolerable discomfort. In addition, data will need to be gathered to determine the effect of turbulence on structural loads to assess the impact on component fatigue life. Put together, these requirements call for testing carried out in actual turbulence. The obvious problem here is that test conditions are difficult to predict and therefore plan for. A major exception here is the turbulence created by steady winds blowing over and around obstacles, buildings and a ship's superstructure and the general effects on flight operations in the near-earth or near-ship environment. Calibrated facilities could play a very useful role here. The effect of discrete gusts can to some extent be simulated with pilot test inputs and this is an established method of testing. Automatic control inputs would be much more effective in this regard as they could be easily 'programmed' to represent a wide variation of gust spectra including the design cases. What is currently lacking is a meaningful database of turbulence characteristics for many helicopter operating environments, particularly nap-of-the-earth flight.

3.3.4 System identification

During the flying qualities testing phase of a development test programme on a new helicopter the occasion can arise when an unexpected handling problem is discovered that simple analysis and inspection of flight records fails to explain. Examples may be a sudden deterioration in stability for particular flight conditions, stronger than usual cross coupling during manoeuvres or perhaps a loss of control effectiveness in some asymmetric flight state. If not quickly understood and resolved, such problems can seriously impede progress and result in a protracted test programme as various design modifications or flight control system 'fixes' are explored. Many current types operating successfully today have experienced such setbacks and although our knowledge base is improving all the time, it is unlikely that future configurations, built to tougher requirements, will be free from their unexpected problems. The first stage in a rational course of investigation would be a return to the simulation model from which the design evolved. A direct comparison of flight and simulation results in the troubled area will reveal the character of the differences but, without further flight test and analysis, it is unlikely that direct time history comparisons will be able to shed much light on the source of the differences. What is required is a more systematic method of comparison that can isolate the modelling errors and identify the required improved representations. This should point, in turn, to the physical origin of the deficiency in question and guide further flight measurements in search of a suitable design solution.

The methods of system or parameter identification described below have been developed partly to serve this objective. The fundamental aim of the methods is the derivation of a mathematical model structure and the estimation of associated parameters that gives the 'best' fit to the flight data. The meaning of 'best' in this context depends upon the minimising criteria used. In general, the most reliable parameter estimates come from data fitting that is optimum in a probabalistic sense rather than based on some timewise error function, *ie* a particular set of derivative estimates are the 'most likely'. Estimation methods in use range from simplistic methods that take no account of measurement or process noise to 'advanced statistical' methods that try to account for both. No attempt is made here to describe the techniques in any great detail; the reader is referred to AGARD Lecture Series LS-104 (Ref 48) for a comprehensive review of the subject.

Nearly 30 years of development has brought the methods of system identification to the point where, in the hands of the specialist, they are providing satisfactory results

for a range of fixed wing aircraft applications. For helicopters, although a limited number of studies have provided encouraging results, it must be said that reliable, foolproof methods are not yet established for a number of reasons. These are reviewed in Ref 49 and summarised below:

a. system complexity - coupled longitudinal/lateral dynamics, interaction with main rotor dynamics,

b. high vibration environment - reducing signal to noise ratio,

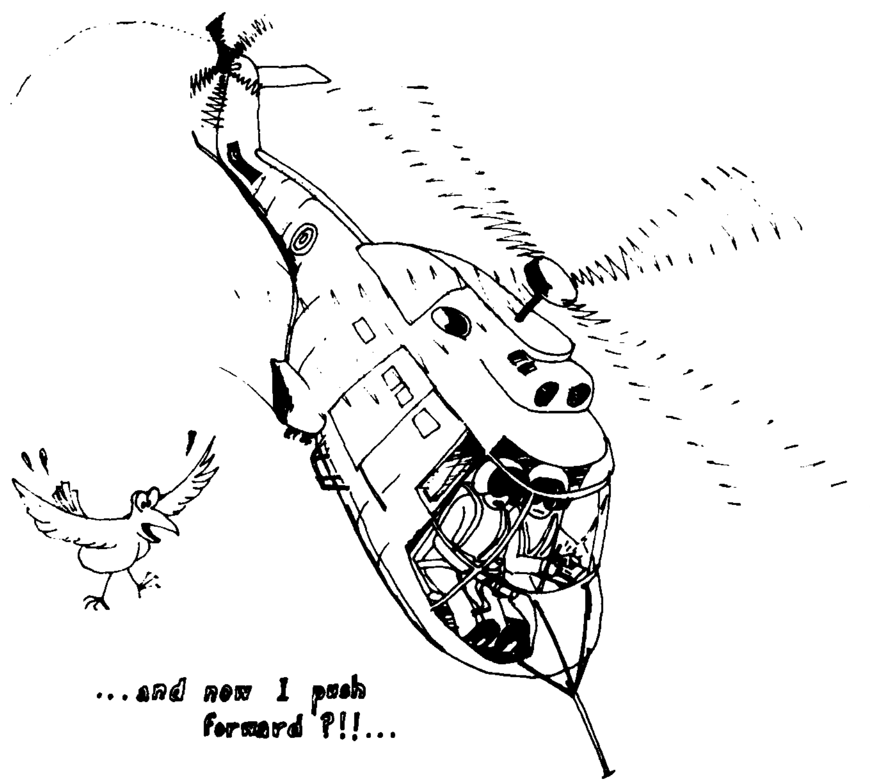
c. instabilities - restricting data record lengths, difficult to trim,

d. nonlinearities - moderately sized inputs required because of (b) leading to airspeed/sideslip/incidence nonlinear effects - process noise,

e. airdata measurement systems sensitive to rotor wake and fuselage flowfield effects - process noise.

Taking account of these special difficulties, a number of integrated approaches have been developed for helicopters (Refs 50-52). The required stages derived from these experiences are summarised in Fig 46 and form the basis of the method under development at RAE Bedford.

The optimal control input design stage aims to define the control input shapes and sequences required to excite the aircraft in such a way that particular response variables and hence their associated parameters are dominant. The design can be performed in the frequency domain based on a bandwidth criterion or more directly through minimisation of parameter estimate error variances (Ref 48). The design will initially be based on results from a simulation model of the aircraft but can be updated as flight data is gathered, if required. Signal shapes that have been used in practice range from complex multisteps to combinations of periodic functions. The requirement for a wide excitation bandwidth along with simplicity led to the DFVLR '3211' multistep illustrated in Fig 47, along with other typical multisteps. These types of input are easily 'flyable' by the pilot but with more complex input shapes it is desirable to apply the controls automatically. A convenient way to achieve this is through the limited authority actuators normally driven by the automatic flight control system (Ref 53).



Following the flight trial a preliminary interpretative and comparative check on the results should be made. This should cover repeatability (three runs at each condition are desirable), linearity (excursions in response, effect of control input size) and enable rough estimates to be derived of damping, frequencies, control sensitivity and the levels of cross coupling involved. Comparison with the simulation model will provide clues regarding the modelling deficiencies, *eg* are nonlinear effects reproduced, frequencies similar, cross couplings correct etc.

The next stage in the process highlighted in Fig 46 involves the computation of filtered or smoothed state estimates from the measurements. With the aircraft kinematic equations providing the system 'model' in an extended Kalman filter/smoother algorithm, calibration factors can be corrected, unmeasured states estimated and the levels of measurement noise on the data reduced. The technique is particularly sensitive to the assumed process noise statistics, or the uncertainty in the validity of the kinematic system model. Atmospheric disturbances and unmodelled degrees of freedom will contribute to the process noise.

Model structure estimation, or the estimation of which degrees of freedom are contributing to the dynamic motion, can be performed in the time or frequency domain, hence the option of a fast Fourier transform (FFT) in Fig 46. Frequency domain estimation has the advantage that the model fit can be restricted to a limited frequency range, enabling reduced order models (*eg* short period or dutch roll modes) to be more easily derived for either single input/single output transfer functions (Ref 54), or multi-degree of freedom models (Ref 55). Least squares step-wise regression algorithms are computationally efficient for multi-variable equation error type analysis. The process suffers from the 'non-uniqueness' problem that generally plagues system identification, in that the fit error continues to decrease as more variables are accommodated, irrespective of their physical significance. Additional criteria need to be used, including the F-ratio test (Ref 56) and confidence level test (parameter covariance matrix) to help judge when an optimum fit has been obtained. Information derived from separate analysis, *eg* static stability measurements, time-vector analysis (Ref 57), can be used at this stage to refine the model structure. Finally, test data from different control input sequences can be used to check the predictive capability of the derived model.

The parameter estimates obtained from the model structure evaluation stage can finally be used as starting values for the maximum likelihood process which iterates towards unbiased, minimum variance values, and provides information on the reliability or uncertainty level of the estimates (Ref 58).

In Fig 46 the use of a simulation model to validate the methodology is stressed. The effects of measurement noise, degrees of freedom, nonlinearities and record length on the various processes can be explored in a controlled fashion and later used to guide the decision making logic when processing flight data.

Major improvements in the reliability of system identification methods for helicopter applications are required before they are available for routine use during flight test development. Although successes have been claimed, the greater number of published cases where dampings have been grossly underestimated and cross couplings badly predicted suggests that considerable specialist effort is required to overcome its use with problems peculiar to helicopters. This is a research task and more fundamental studies with elements of the structured approach described above are urgently required.

3.3.5 AFCS development flight testing

Operational requirements for most modern helicopters demand that some form of automatic flight control system is installed to assist in the flying task. Systems can range from simple stability augmentation systems (SAS) to sophisticated stability and control augmentation systems (SCAS) tailored over a range of flight conditions and supplemented with a variety of autopilot functions. Such systems usually interface with the primary mechanical flight control system through limited authority series actuators and rate limited parallel trim actuators. The flight test requirements for these systems fall into two categories.

(a) Tests to optimise system performance and handling qualities.

(b) Tests to establish the effects of failures.

These tests will normally be performed by the manufacturer, in liaison with the avionics sub-contractor, during development flight testing if the system has been designed as an integral part of the vehicle. The optimisation phase will have as an objective the clear demonstration of compliance with the appropriate set of requirements from both aspects of system operation and handling qualities. It will therefore need to encompass the complete range of flying qualities tes s even though these may have already been carried out on the basic aircraft. Any augmentation saturation effects, for example, will need to be carefully explored. In addition, tests will be required to determine system performance for any special features, *eg* control system mode changes (heading hold to turn coordination) and automatic approach, hover and departure autopilot function on the SH-60B (Ref 59). For in-service aircraft there is often a requirement to extend their capability towards 'all weather' operations and additional testing will be needed for avionics that contribute to this aim (Ref 60). Control system optimisation testing can, of course, highlight problem areas as control system and airframe modes interact in unexpected ways. Ref 44, for example, describes how, with the AFCS engaged, vertical vibration was aggravated during recovery from high speed autorotations with the SH-60B. In this case, 'relocation of the SAS pitch gyro to the first fuselage bending mode anti-node position produced the desired improvement'. Comprehensive instrumentation is usually required to assist 'trouble shooting' during this type of testing and both aeromechanical and avionics disciplines are involved.

The limited integrity of single lanes of an AFCS demand that a detailed failure modes and effects analysis be performed and additional flight testing carried out to

establish safe recovery techniques and provide practical evidence of the flight régimes where pilot recovery is possible. Tests need to be carried out with the appropriate intervention times made up of the aircraft response time (time for appropriate cue to build up) and the pilot response time. The intervention times that have to be complied with depend on the certification authority. The UK military requirements (Ref 2), for example, refer to active, attentive and passive flight segments with varying pilot response times. This reference also advises on the test techniques to adopt when simulating system failures. Types of failure range from slow to rapid runaways to oscillatory malfunctions. Testing requirements will depend upon the intended aircraft role and the flight segments within these. For example, Ref 2 suggests that for active flight segments runaways should be injected without warning the pilot while for attentive and passive flight segments the pilot should be warned. In general, a fairly thorough test programme will be required to cover clearance, with particularly detailed and careful tests at the edges of the flight envelope. Most certification agencies allow an extension to the normal flight envelope boundaries for the recovery phase following a failure, *eg* Ref 7 states that 'during high speed malfunction testing the maximum speed allowable during malfunction or during recovery is $1.11V_{ne}$.

A simple qualitative assessment of AFCS runaways adopted by the Empire Test Pilot School (Ref 8) as part of the training programme, and suitable for preview trials is described as follows:

> '...The first runaway is initiated at V_{mp} with actuators null (if applicable), with no delay before pilot intervention. The runaway envelope is opened incrementally, increasing/decreasing speed, displacing actuators, and increasing delay time until either the defined intervention times are achieved or the pilot encounters a limitation. The runaways must be carried out in both directions, in all channels, if possible into the extremes of the flight envelope in terms of IAS and manoeuvres. The data available from such a trial would be the delay time, the maximum height loss and the proximity to other limiting conditions over the entire envelope. Finally some runaways would be carried out without warning in all axes and, if necessary, runaways would be carried out while flying on instruments...'

The key to establishing flight clearance from failure testing is to determine where safe recoveries can be made without exceeding aircraft limits and to check that clear unambiguous cues attract the pilot's attention to the failure, and that the recovery technique is natural and obvious. Allowable attitude excursions during failure and subsequent recovery vary from type to type and are based largely on the self-correcting characteristics of the aircraft combined with the immediacy of the pilot cue and subsequent control response.

Flight testing for pitch runaways on the Westland 30 helicopter revealed possible excursions into retreating blade stall just beyond the level flight airspeed boundary (Ref 61). At lower speeds it was generally found that the aircraft would naturally self correct following a single lane pitch runaway and the pitch attitude transient remained below 30° without pilot intervention. At some critical speed the pilot noticed that the severity of the runaway increased and cyclic intervention was required to keep the pitch excursion below 30°. Analysis of data from the instrumented aircraft revealed that limited retreating blade stall occurred as the rotor disc incidence increased. Fig 48 shows the airspeed envelope for the W30-100 aircraft and the test points covered during the runaway tests.

Full circles represent the conditions where intervention was required as a result of rotor stall (rapid runaway). The motion and associated vibration cues during these rapid runaways readily manifested themselves to the pilot who, with a small forward cyclic input (10%), could easily correct the situation. While the hingeless rotor configuration on the Westland 30 results in somewhat higher pitch rates, it also endows the pilot with the precise and crisp control response necessary to recover in these situations.

Looking to the future, and the potential development of full time, active fly-by-light flight control systems for helicopters, one can foresee a requirement to integrate more closely the system development and performance and flight envelope testing. New test techniques and procedures will need to be adopted and, with the possibility of several levels of reversionary modes available, failure testing will need to be re-appraised. The consequences of control limits being reached or flight limitations being exceeded without the pilot being aware of the situation could be distrastrous, which implies another level of active monitoring. High integrity will be required from these systems and if the promise of uncompromised safety and performance is to be realised, a fresh approach to testing, based on experience with fixed wing aircraft, will need to be debated in the near future.

3.3.6 Role-related flying qualities testing

The clinical test techniques described in previous sections provide the raw material from which the degree of compliance with handling requirements can be established. These test results, though necessary, are not sufficient however for demonstrating an aircraft's suitability for a given role. This same point was made in the Chapter on Performance Testing (see 2.3.2) and is even more true for flying qualities evaluation. In fact, the close interaction of flying qualities and performance when checking role-suitability requires that both aspects be evaluated together: task performance becomes a function of vehicle and pilot performance. This final section is

concerned then with pilot performance and reviews a range of new test techniques under development.

At this stage of testing the subjective impressions of the pilot will be all important. He will now be able to see more clearly the consequences of all the features exposed during clinical testing. He will be expected to quantify his impressions using a rating scale (Ref 62) and suggest the kind of improvements that are essential or desirable.

The need for a flight phase dedicated for role or mission testing has led in turn to the development of mission oriented criteria, based on stylised task elements. Reference 63, for example, describes flight experiments to identify flying qualities limitations for NOE flight; tasks included the slalom, teardrop and 'S' turns and lateral quick stop manoeuvres (Fig 49). Results from this and other studies in the 1960's and 70's began to form a database of flight results that could be used to establish desirable handling characteristics for future designs. By flying different helicopters through the same task elements, a wide range of handling issues could be assessed in an operational context, including primary response sensitivity and damping, cross couplings, manoeuvre stability and rotorspeed control effects. By the mid 1970's the US were in a position to specify mission oriented criteria for their UTTAS and AAH requirements (Ref 64), and later the LAMPS Mk III (Ref 65). Looking to the future, the proposed Mil Spec 8501 revision (Ref 31) suggests more than 60 task elements to be included in the 'general requirements' with detailed task performance quantified for each specific requirement. Within the new structure proposed in Ref 31, the concept of the usable cue environment (UCE) is introduced to further categorise handling criteria based on different levels of outside visual cues and artificial vision aids available.

The need for combined mission performance and handling criteria demands that equal attention be paid to both pilot and task performance and has initiated a number of studies aimed at correlating task workload and performance. The level of pilot workload for a given task will depend on the aircraft handling qualities in the prevailing atmospheric conditions, the task cues and aircraft performance itself. A pilot will generally attempt to adopt a control strategy that maximises performance while minimising workload. This must involve a compromise, and depending upon the consequences to the mission, one or other will usually suffer. The key to understanding how different pilots cope with this compromise, how task performance and workload correlate and how sensible criteria for task-related flying qualities can be constructed, lies in the identification of pilot control strategy. The studies described below rely on a combination of thorough debrief sessions aided by questionnaires and the use of pilot rating scales, and a detailed analysis of pilot's controls and task variables in the time and frequency domain.

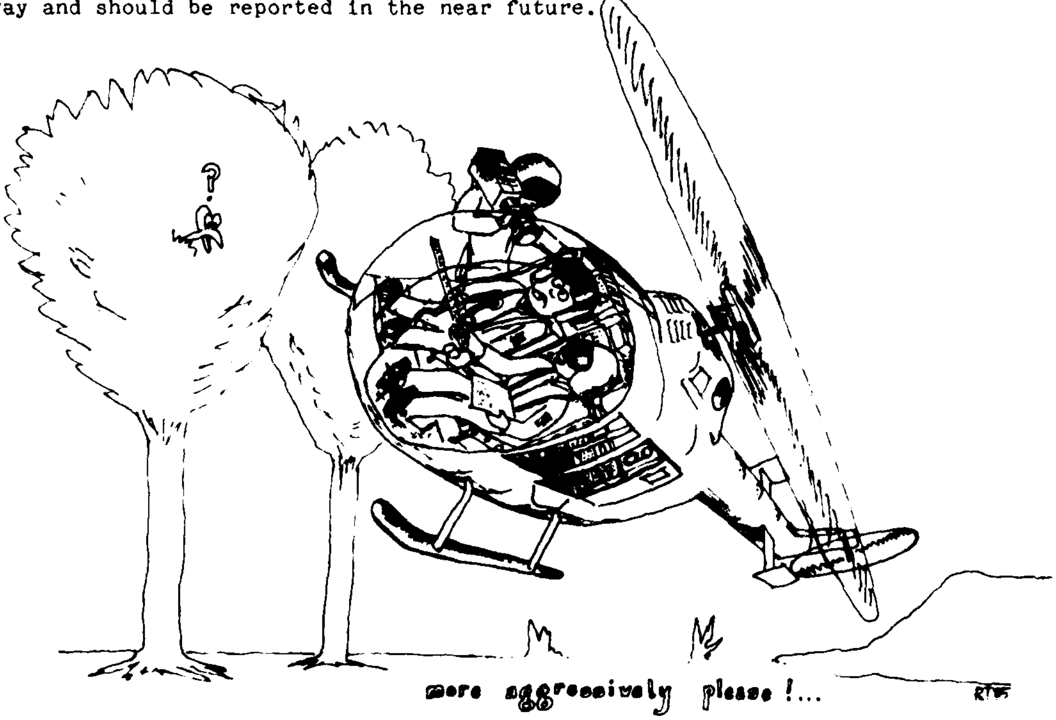
A technique already in use during flight test development for fixed wing aircraft at the US Air Force Test Center, Edwards Air Force Base, aptly named SIFT (System Identification From Tracking), is described in Ref 66. The analysis technique in SIFT has been developed, largely through frequency domain methods, to highlight handling qualities that affect performance during closed loop flying tasks. Multi-variable frequency analysis enables individual loop control strategies to be isolated and interpreted across a defined bandwidth. A major claim of the technique is that by requiring the pilot to perform his task 'aggressively' and reducing errors to a minimum, the full range of handling deficiencies, including pilot induced oscillation (PIO) tendancies, can be explored and identified in the frequency analysis. Ref 67 describes the use of the SIFT technique to identify the cause of PIO's experienced by CH-53 pilots during the approach when carrying large underslung loads. Three incidents when emergency release of the load was required are reported and explained through a frequency analysis of the collective -normal acceleration loop; pilot interaction with a weakly damped helicopter/load mode causing the instability. Task demands will not always require the pilot to adopt an aggressive control strategy, but on the basis that this type of flying technique will represent a 'worst case', they should be included in a test programme to expose potentially 'hidden' handling deficiencies.

The relationship between control strategy and piloting workload can only be properly addressed through an understanding of task cues. Even when the full range of visual flight status cues are available, a pilot's workload can increase through the need to track aggressively (increase gain), anticipate the response (increase lead time) or divide attention between task variables. The recognition of these effects has led to considerable development of manual control theory for piloted vehicles and the use of pilot models to describe control strategy and establish workload criteria. One such development that is currently finding useful application in the helicopter field is known as the non-intrusive pilot identification procedure (NIPIP) (Ref 68). The emphasis on non-intrusive measurement is based on the premise that (from Ref 68)..."Our dependance upon pilots to assess their own actions, perception, and degree of stress, however, can interfere with what it is that we are trying to measure". The reference further argues that it is difficult for pilots to quantify their control strategy or describe cue patterns used. While it is true that pilots rely on instinctive skills built up through training and experience so that they may not be conscious of their detailed control activity, the role of subjective impressions should not be undervalued. They can be used to support and guide analysis which can then, if properly conducted, shed considerably more light on piloting techniques. One of the basic tools of NIPIP is a running least squares estimation algorithm that can be used to identify, in the time and frequency domain, parameter values for a derived closed loop pilot/task cue model. The method can be applied to time varying as well as 'stationary' tasks provided the model parameters do

not vary so rapidly that the data windows are too small to provide meaningful results. This does not, however, appear to be a serious restriction. Ref 69, for example, describes an application of the method to identify control strategy for a range of NOE tasks, with detailed results presented for the bob-up and hover turn manoeuvres. Task performance and pilot control strategy are evaluated for both flight and piloted simulation results for a UH-60 helicopter in a series of simulation validation experiments. Simulator fidelity is measured by the degree to which the visual and motion cues and aircraft dynamics induce an equivalent control strategy to that adopted in real flight. Although the bob-up manoeuvre clearly requires several rapid changes in control strategy, a successful analysis was performed by dividing the manoeuvre into three distinct elements - a transition phase, a rapid response phase and an error reduction phase. Anomalies between flight and simulator results, highlighted by pilot impressions, were exposed in the analysis as different pilot model structures were identified in the three phases for the two cases. Phase disparities between simulator motion and visual cues were considered the most likely source of the anomalies and results from the analysis can be used as a basis for improved cueing criteria. NIPIP techniques are proving a valuable tool in control strategy analysis and should serve well the substantiation of new handling criteria.

Flight test techniques for establishing task performance criteria and control strategy have also been developed at the DFVLR, Braunschweig (Ref 70). Once again, tactical helicopter operations are receiving attention and Ref 70 reports results for operational standard BO-105 and UH-1 helicopters in 'dolphin' (hurdle hopping) and 'slalom' flying trials. Criteria based on kinematic characteristics of the manoeuvres and first order statistical analysis of control activity are proposed. For the slalom task an on-line analysis technique has been developed to compute the task errors and derive a 'score factor' for the run. This provides a good appreciation of how far the pilot has advanced along his learning curve for the aircraft configuration being flown.

The need for more precisely defined tasks to promote aggressive closed loop flight path tracking has led to the creation of a set of ground marked courses at RAE Bedford. These include a series of concentric circular tracks, a spiral and triple bend course. Identical tasks have been marked on the simulator model terrain to aid control strategy validation work. The circles provide an essentially 'stationary' type of task, at least in low wind conditions; the spiral a progressive non-stationary task and the triple bend a combination of stationary and rapid transition (roll reversals) elements. Task variables include track over the ground, height, speed and balance and a variety of different control strategies have been explored to isolate the principal workload ingredients -concentration, anticipation, divided attention and spare capacity. Detailed questionnaires have been designed to help the test pilots describe the interaction between task cues, control strategy and task performance. Fig 50 shows the power spectrum of lateral cyclic control for one pilot flying a medium circle (515' diameter) at 80 kn in a 10 kn wind. Four different control strategies are considered, cyclic only (S1), cyclic + collective (S2), cyclic + pedals (S3) and cyclic + collective + pedals (S4). The low frequency power (below 0.1 Hz) is due largely to wind effects that induce changes in mean bank angle required to hold track from 45° - 60°. The marked increase in control activity for S3 and S4 around 1 Hz arises on the one hand from the use of pedal to maintain balance, inducing rolling moments and strong excitation of the oscillatory dutch roll mode and also on the more aggressive strategy used. The increased effort by the pilot for S3 and S4 strategies is rewarded by much smaller track errors as shown in the accompanying figure. More detailed analysis of different pilot's control strategies is currently underway and should be reported in the near future.



The research activities described above are all aiming to derive reliable assessment methods for use in deriving handling criteria, simulator validation studies and for comparing results, from both ground based and airborne simulators, of the many potential candidate configurations with active control. These and related issues are covered in more detail in other lectures in the present series (Ref 71).

4 DISCUSSION AND CONCLUSIONS

This lecture has attempted to provide a broad coverage of the topic flight testing for performance and flying qualities, concentrating mainly on test techniques and the interpretation of test data. Throughout the paper reference has been made to the different categories of flight testing required during Industry's development phase, Government's compliance demonstration and flight manual production and specialised research activities. Although objectives differ, many of the test techniques themselves are common for the three categories. Both quasi-steady and dynamic testing have been addressed and typical results, derived where possible from actual test data, presented to illustrate current performance and handling characteristics. The author has, of course, been selective both in the topics covered and results presented, based on a mixed criteria of reviewing established techniques in a systematic fashion and introducing new approaches, particularly with regard to role-related testing, where these are clearly required. Some of the topics omitted in the selection process have been highlighted 'in passing' and appropriate references given.

It is relevant at the end of this lecture to look to the future and to select from the various themes covered, the important changes taking place that will impact strongly on flight test techniques and philosophy. First the emergence and publication of new requirements for flying qualities and mission performance will lead to a new and wide range of flight assessment techniques. These are already under development largely at research organisations but need to have proven authenticity before they become useful during aircraft certification. There is an urgency in this task if the evaluation techniques are to be built into the new military specifications. Closely allied to the development of test techniques is the requirement for reliable procedures for analysing test data. These include methods for deriving handling characteristics from clinical tests (system identification), correlating task performance and pilot workload from role-related tests (non-intrusive control strategy identification) and the establishment of vehicle performance criteria for different roles (*eg* agility). The requirements for future helicopters are likely to demand the ability to operate in much harsher environments than currently possible. The database on turbulence characteristics in the nap-of-the-earth is particularly sparse at the moment. This needs to be built up from test data and test techniques to explore the impact on handling and structural integrity improved.

The development of active control technology for helicopters will, if fully exploited, introduce radical changes to the flying qualities and flight envelope boundaries testing philosophy. Optimisation of control configuration during development testing will require strict adherence to procedures and the 'incremental' approach will develop a new meaning. Experience gained from fixed wing aircraft test programmes will be invaluable here.

Finally, it is tempting to contemplate how endeavours to standardise certification test requirements will fare over the next ten years. There are obvious benefits from the adoption of an international standard but individual countries are likely to want to place emphasis on different areas of interest, and preserve some of their proven methods of compliance demonstration. A unified structure should, however, be able to accommodate these differences while at the same time enabling Industry to design for a world-wide market.

The challenges ahead in helicopter flight testing are stimulating but they will not be without risk and test techniques and instrumentation need development to balance this. The wealth of experience and tradition, so necessary for continued safety in this exacting discipline, needs to be built on and expanded for these challenges to be exploited.

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In the production of this lecture special gratitude is felt for members of the Flight Research Division, RAE Bedford who have given invaluable support. Finally, thanks to Phil Brotherhood who applied his considerable experience to the task of reviewing the first draft.

The author, of course, takes full responsibility for any errors or ambiguities the reader may come across.

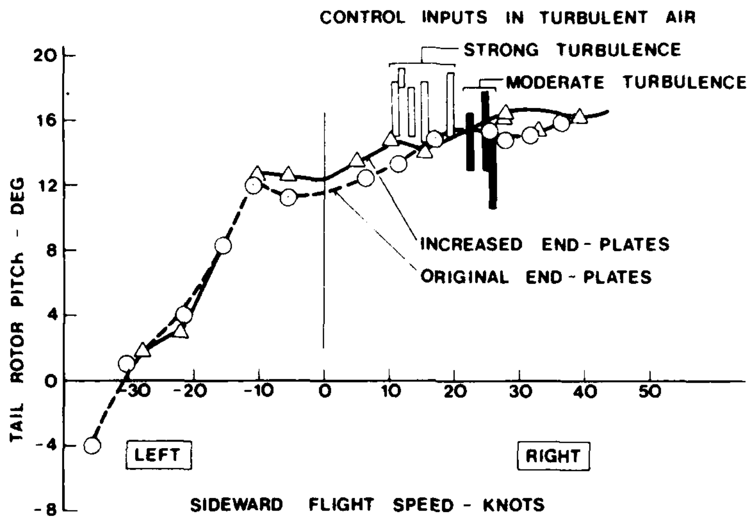

Fig 32 BK 117 Sideflight Trim Characteristics;
Max Gross Weight (from Ref 38)

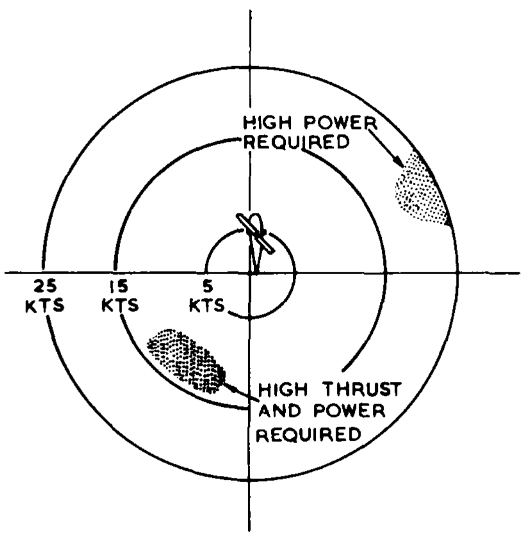

Fig 33 Critical Control Azimuths for Hover
in Wind (from Ref 39)

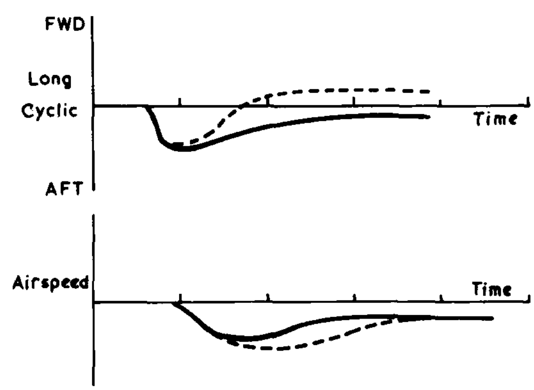

Fig 34 Cyclic/Airspeed Variations for Positive
and Negative Speed Stability

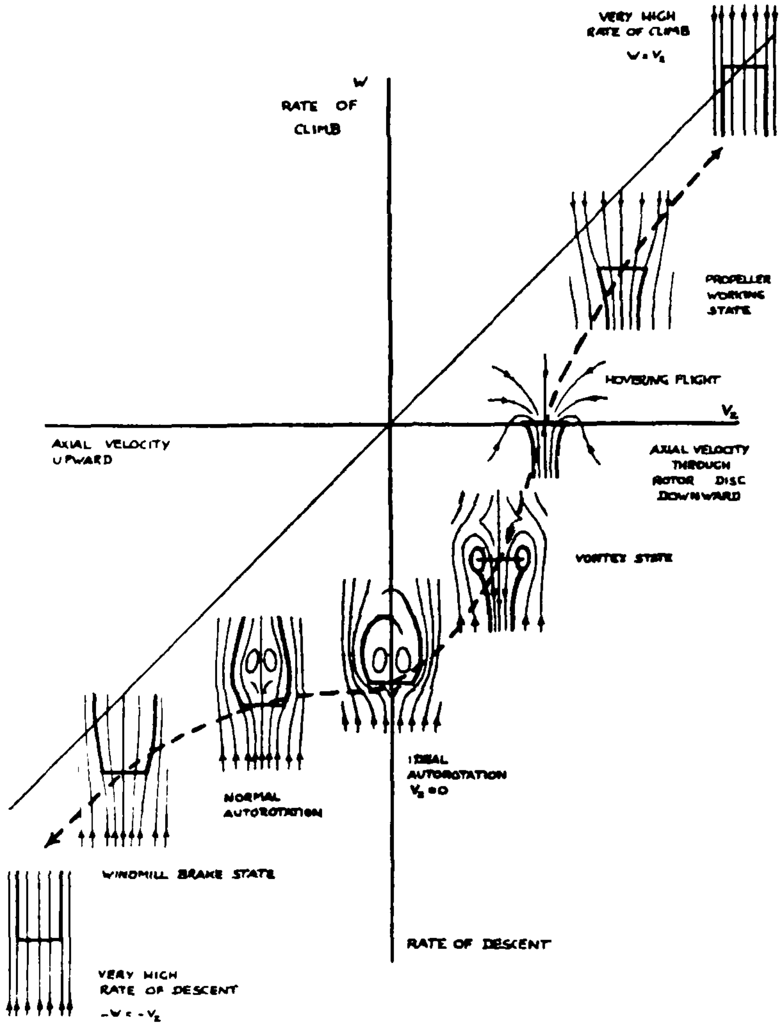

Fig 30 Flow Patterns Through a Rotor in Vertical Flight (from Ref 34)

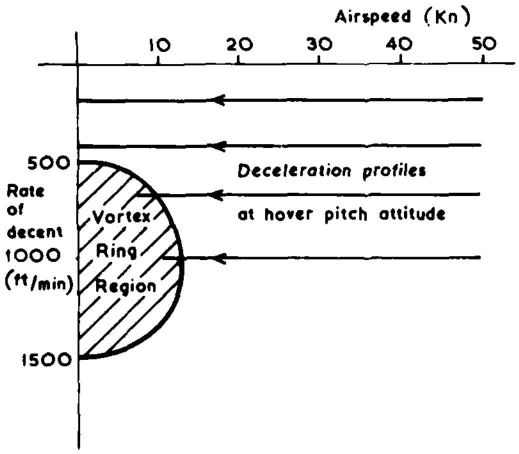

Fig 31 Deceleration Profiles into the Vortex Ring Region

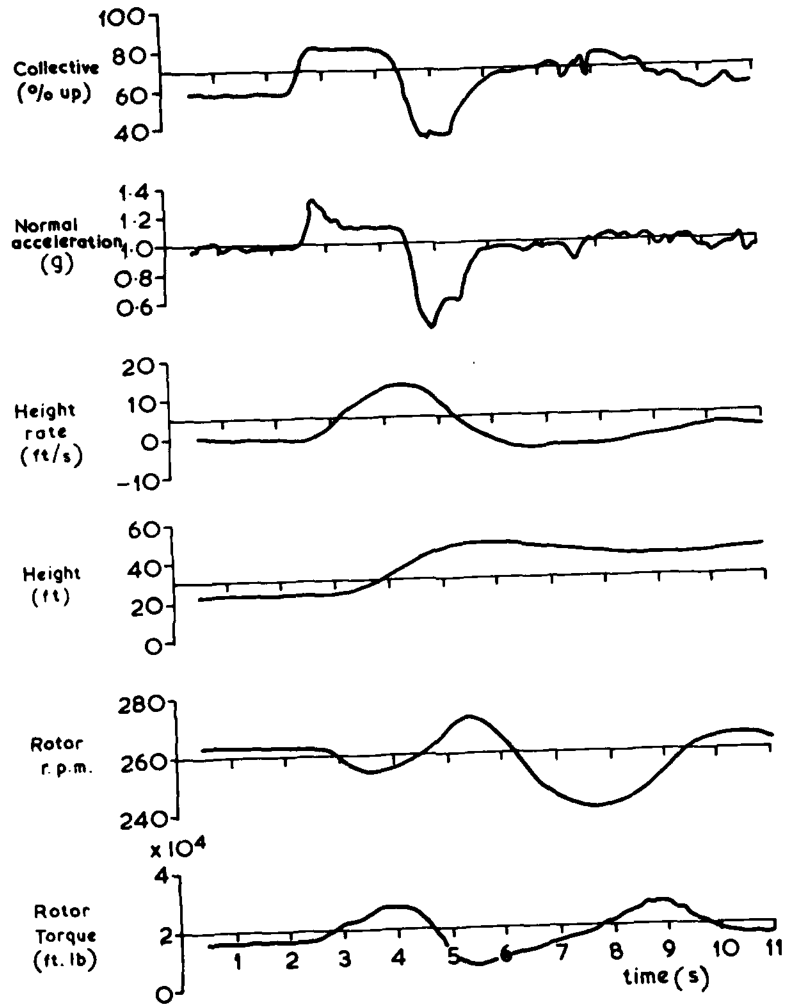

Fig 29 Response Characteristics During 25 ft Bob-up

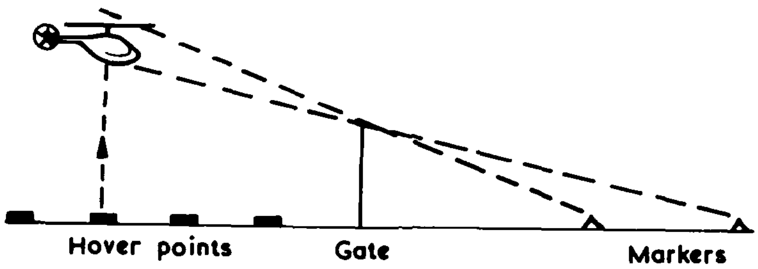

Fig 27 Bob-up Task

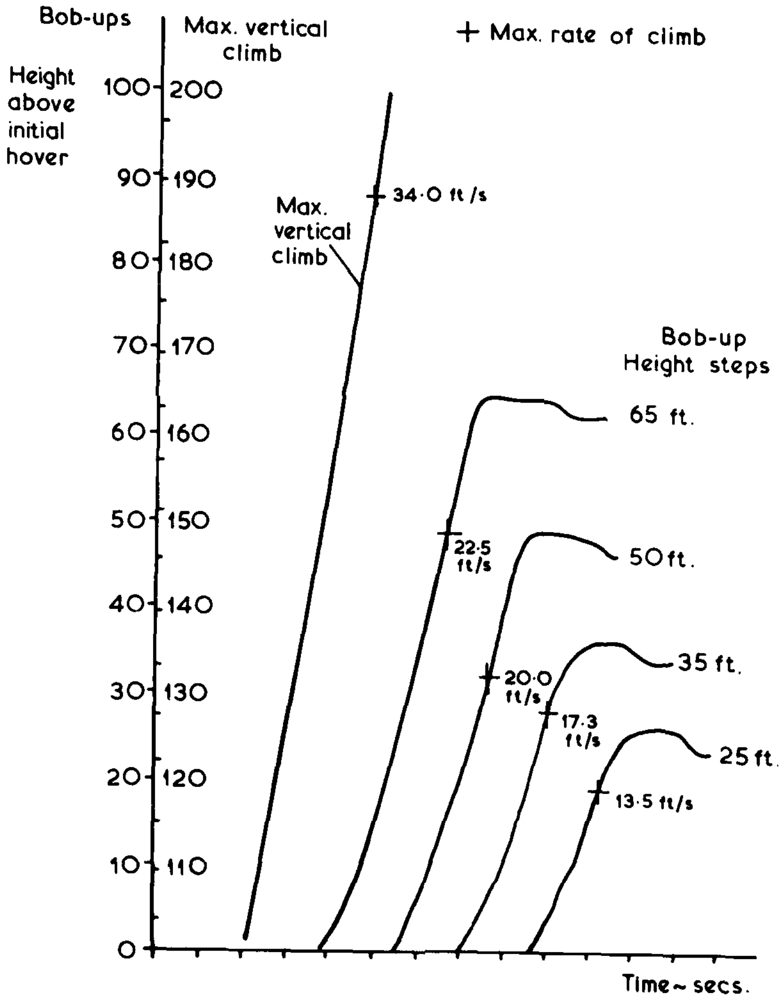

Fig 28 Height Variation During Bob-ups

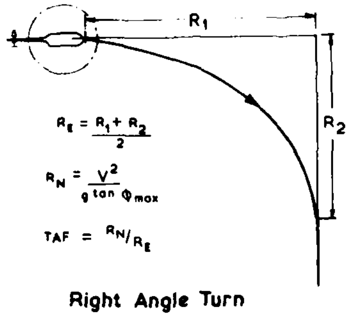

Fig 25 Right Angle Turn Manoeuvre

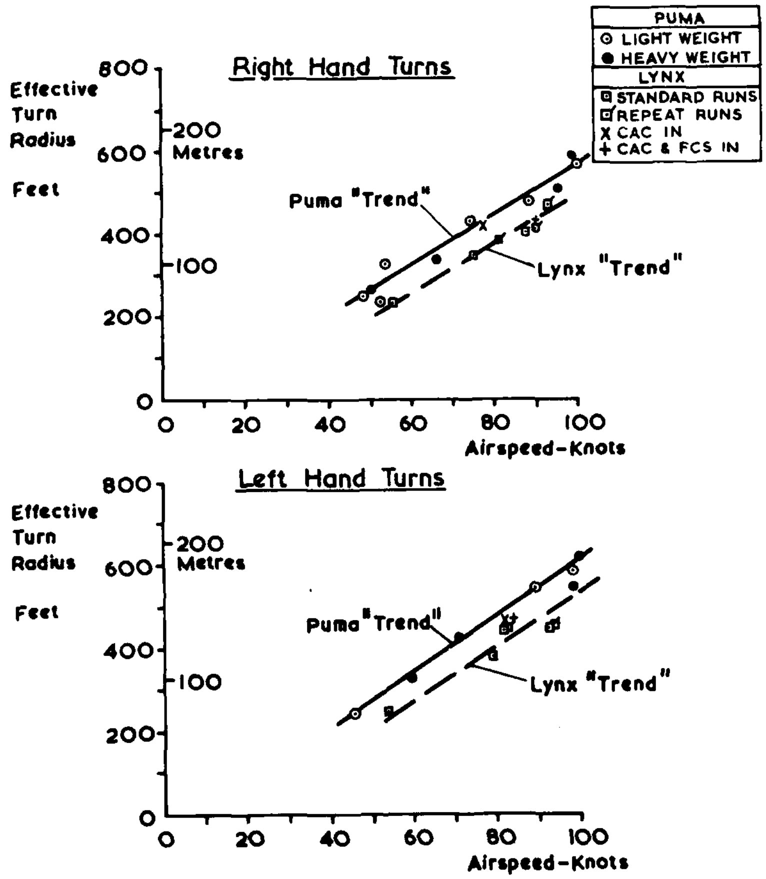

Fig 26 Puma and Lynx Turning Performance for Right and Left Turns (from Ref 32)

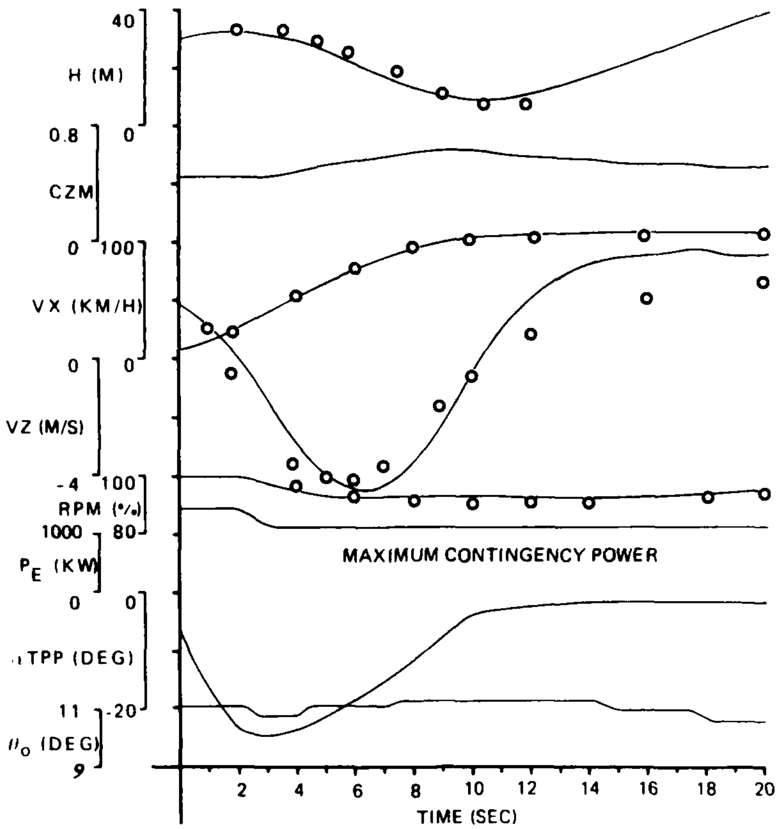

Fig 23 SA 330 - Emergency Procedure; Failure of One Engine During Take-off (from Ref 24)

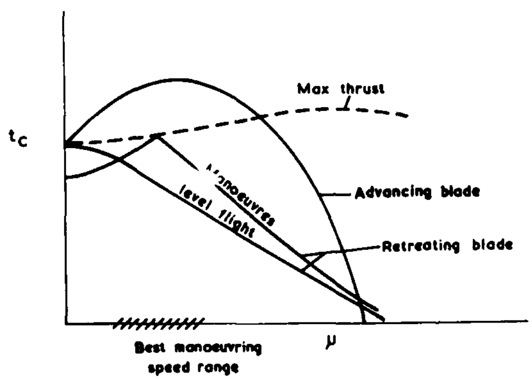

Fig 24 Manoeuvre Performance Compatible with Maximum Thrust

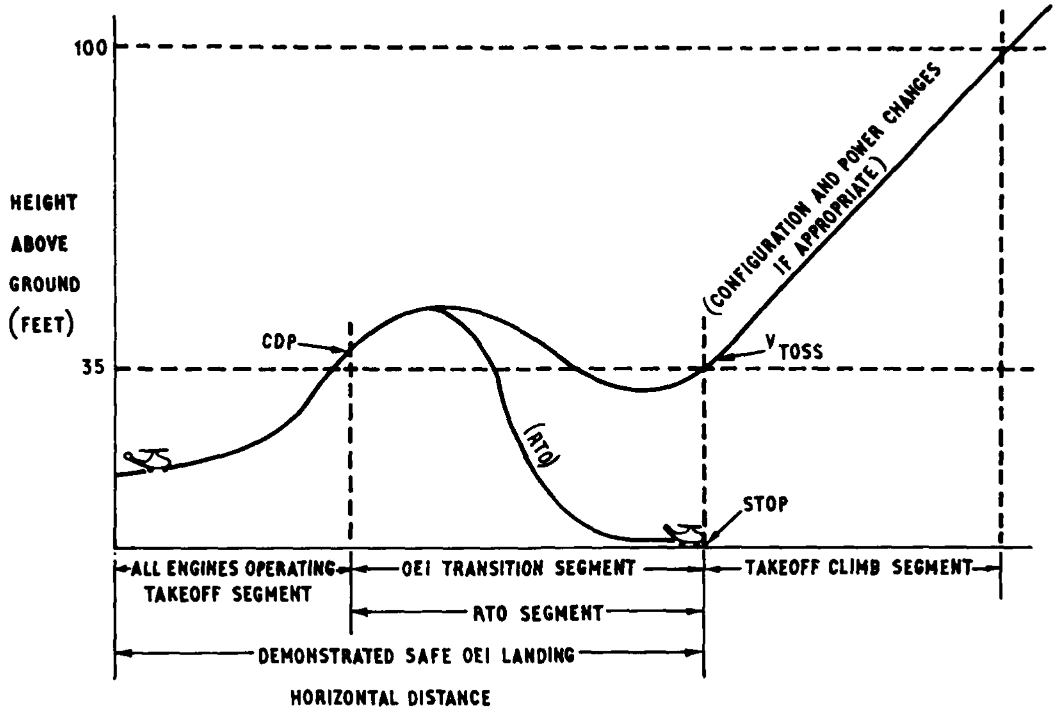

Fig 21 Take-off Performance - Category A (from Ref 7)

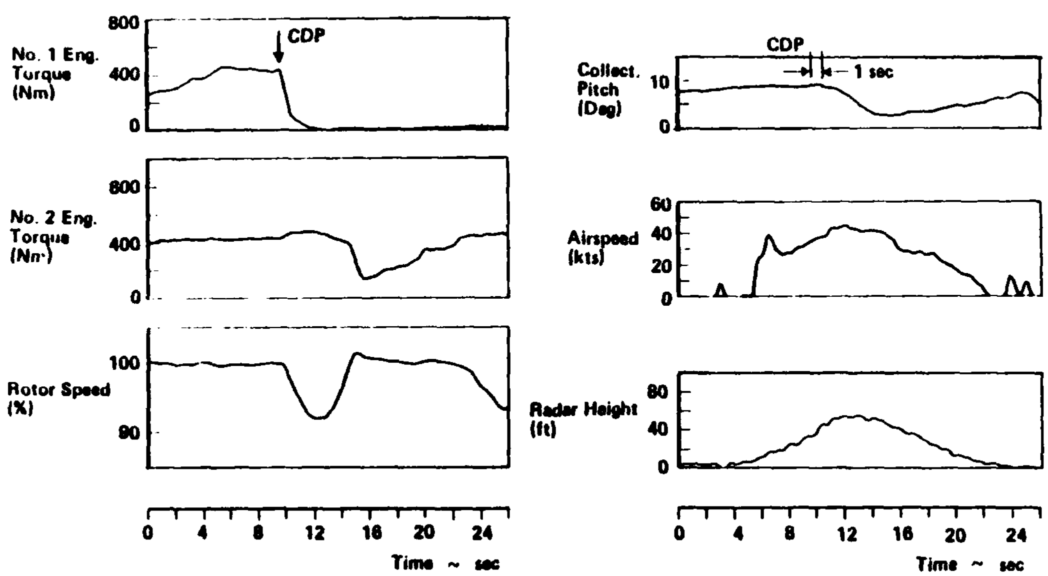

Fig 22 BK 117 - Time History of Category A Rejected Take-off (from Ref 21)

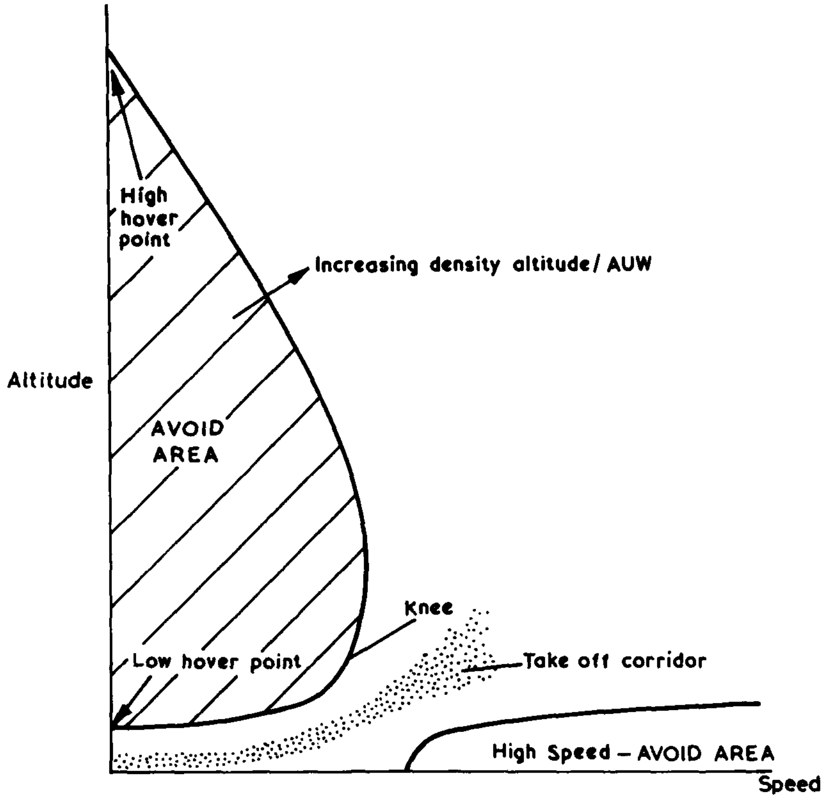

Fig 19 Height-Velocity Diagram Showing Avoid Areas

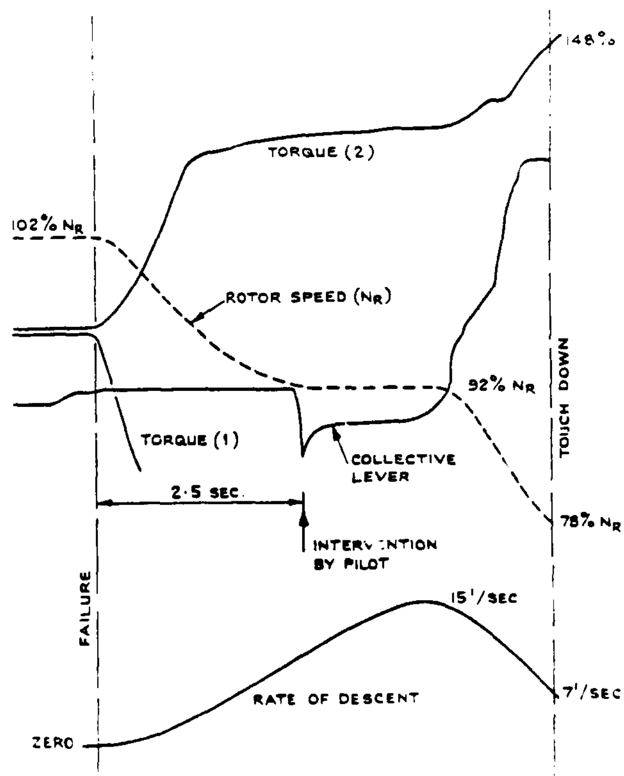

Fig 20 Single Engine Failure from Low Hover Point (from Ref 8)

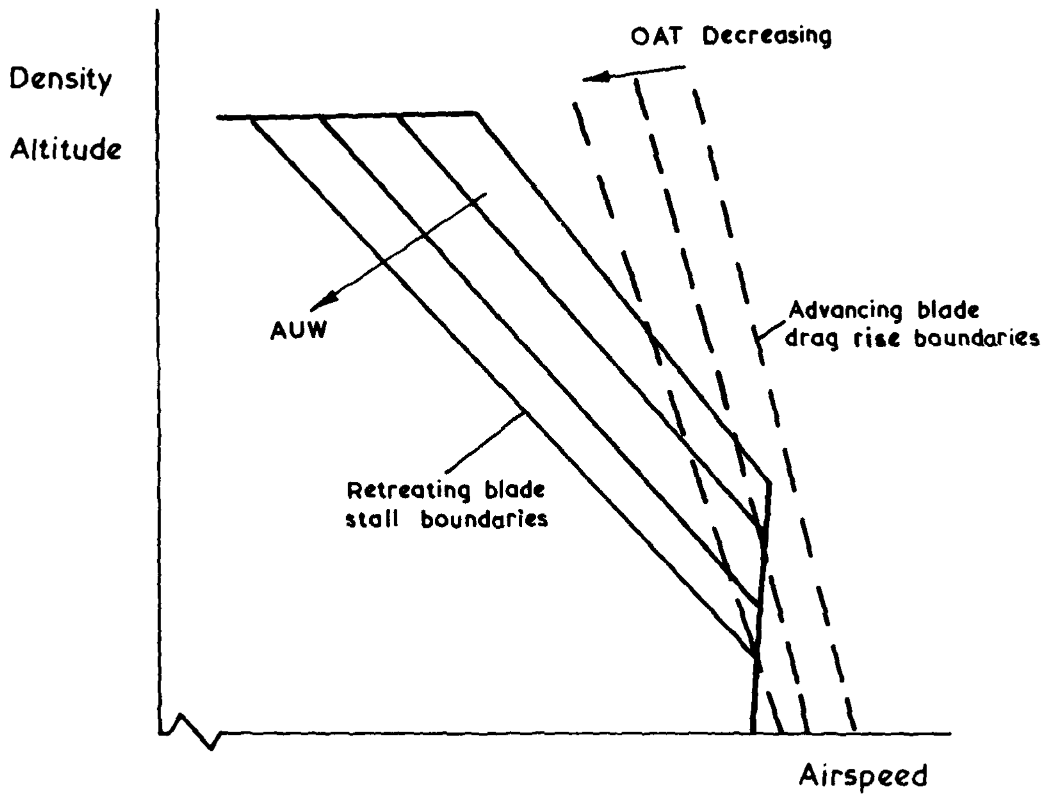

Fig 17 Airspeed Limitations

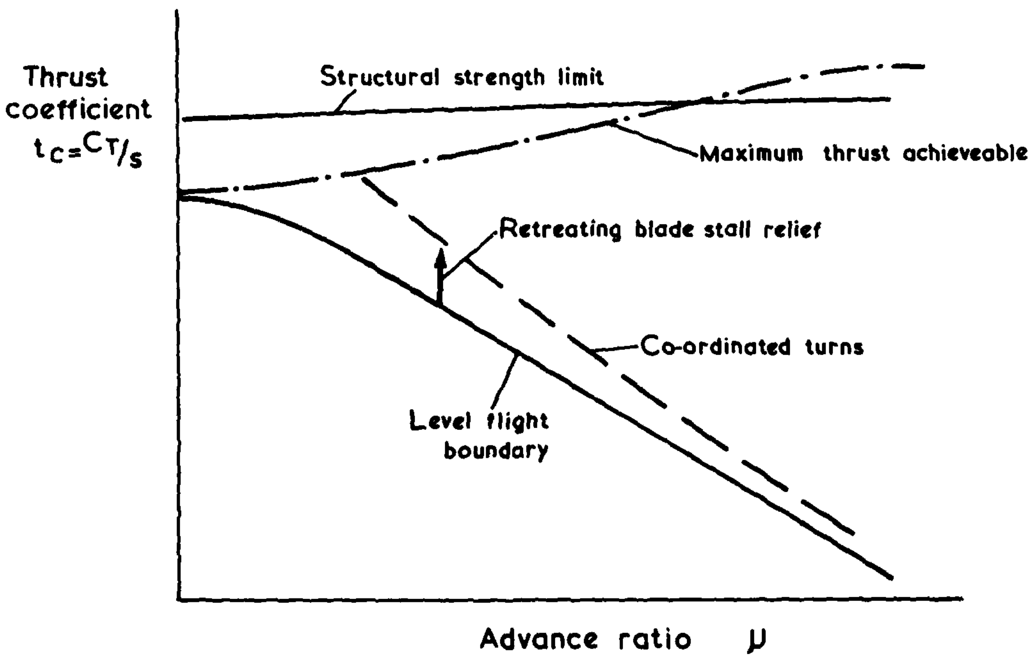

Fig 18 Retreating Blade Stall Boundaries

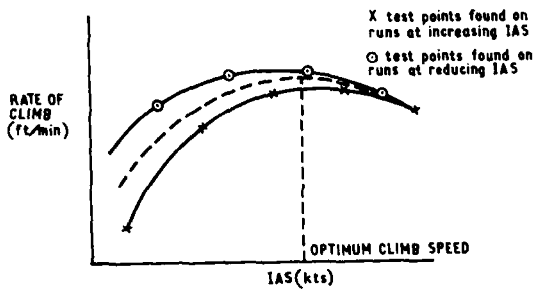

Fig 14 Rate of Climb vs IAS (from Ref 8)

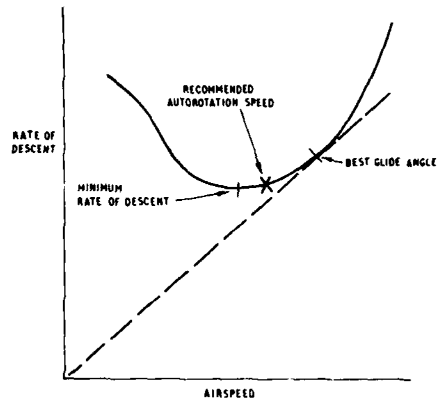

Fig 15 Autorotational Characteristics (from Ref 7)

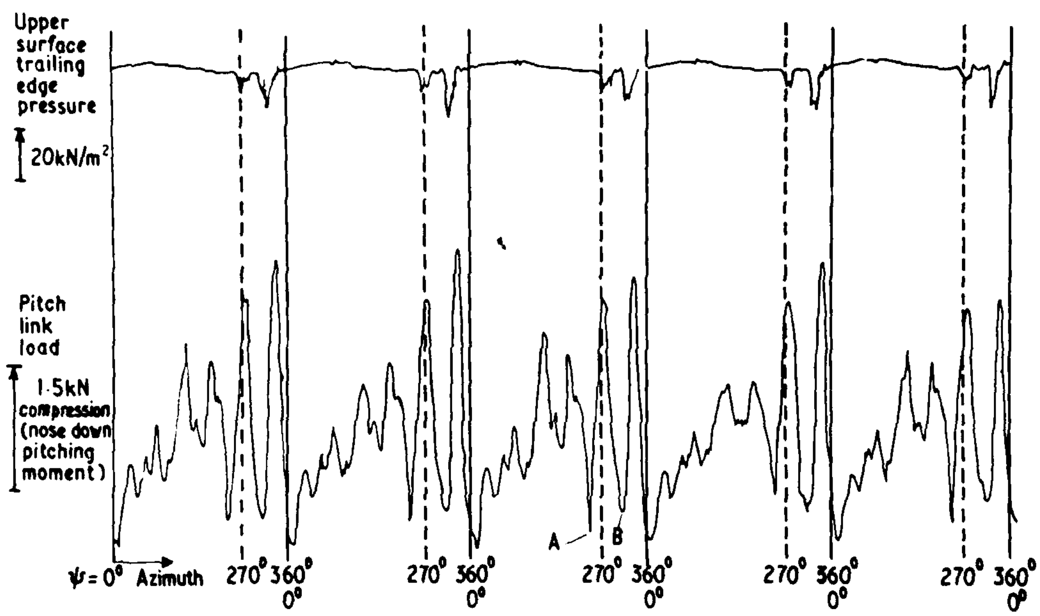

Fig 16 Correlation of Trailing Edge Pressure and Pitch Link Load.
Speed 75 kn. 1.52 m Roughened (from Ref 20)

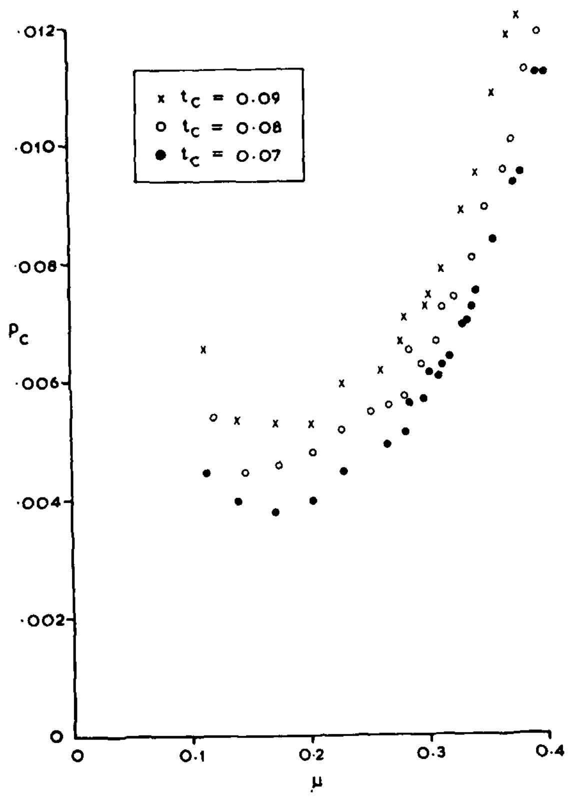

Fig 12 Puma XW 241 - Level Flight Performance Mt = 0.59

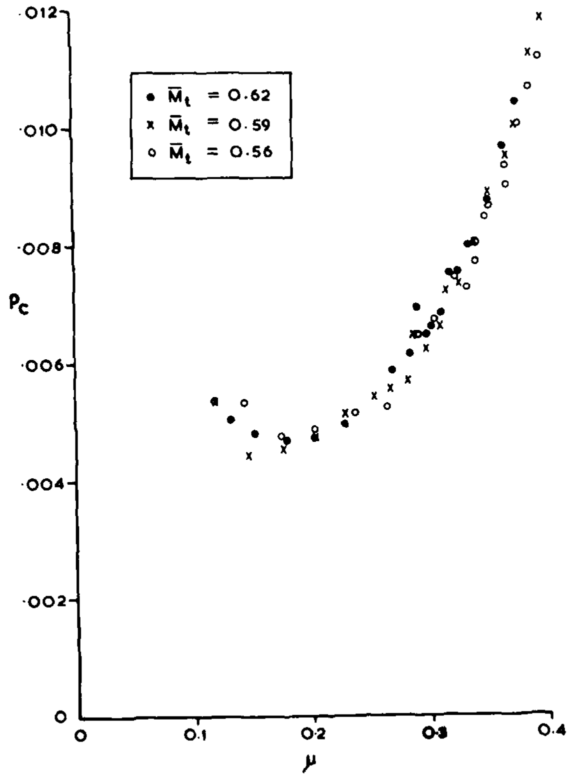

Fig 13 Puma XW 241 - Level Flight Performance t_c = 0.08

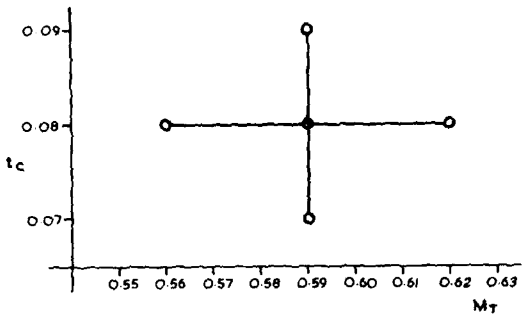

Fig 10 t_c, M_t Test Conditions

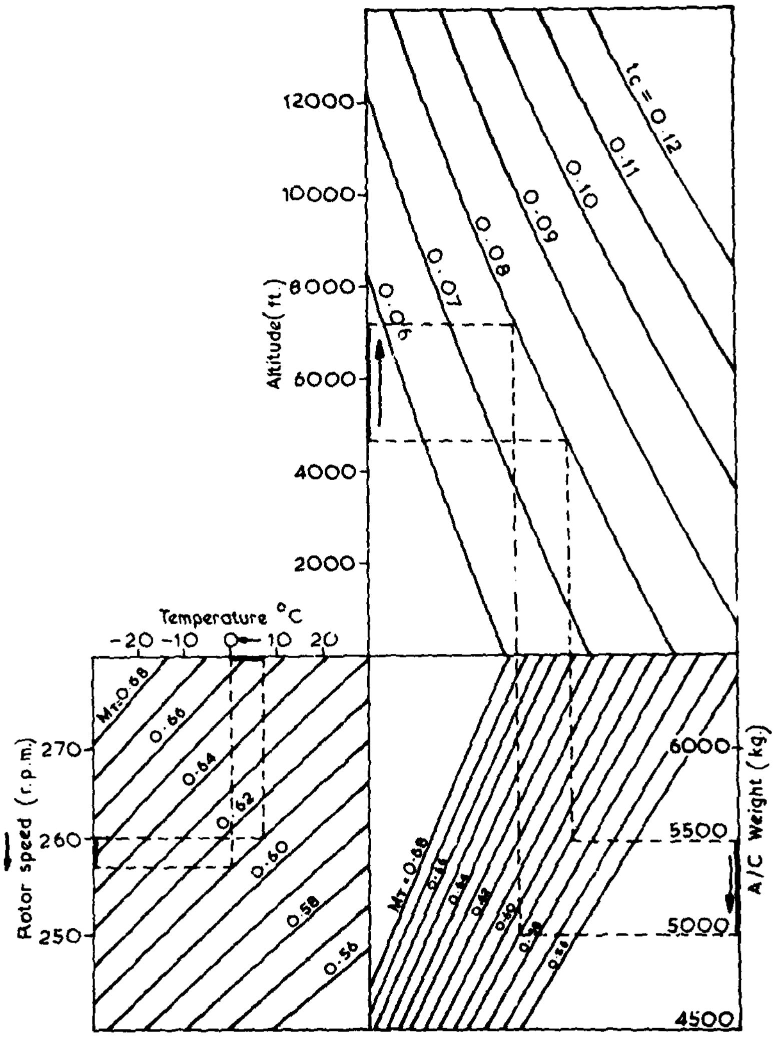

Fig 11 Test Chart for Deriving Constant t_c, M_t Conditions

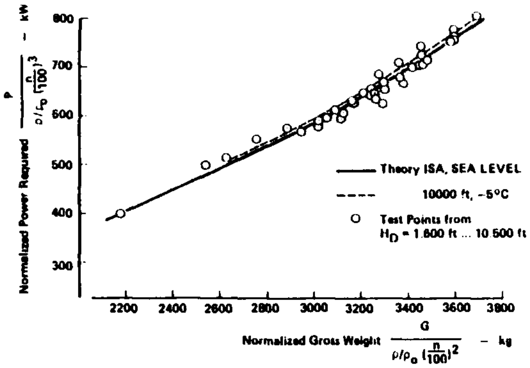

Fig 6 BK 117 - Power Required vs Gross Weight (Hover OGE) (from Ref 21)

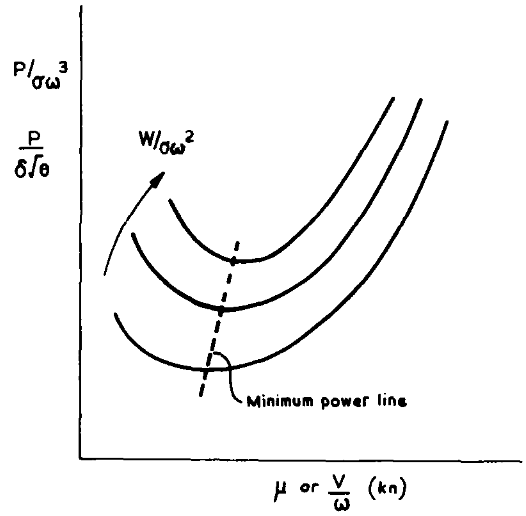

Fig 8 Level Flight Performance

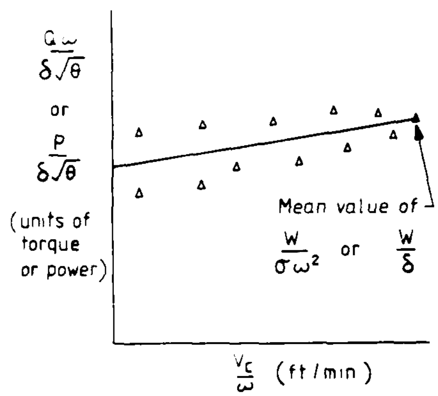

Fig 7 Vertical Climb Performance (from Ref 8)

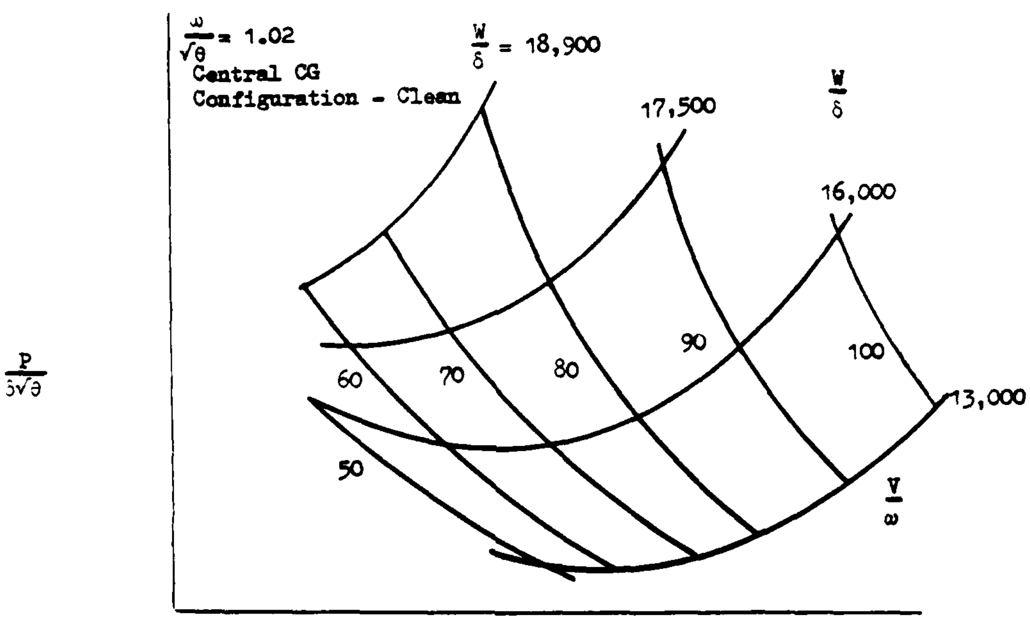

Fig 9 Power Carpet Plot, Level Flight (from Ref 8)

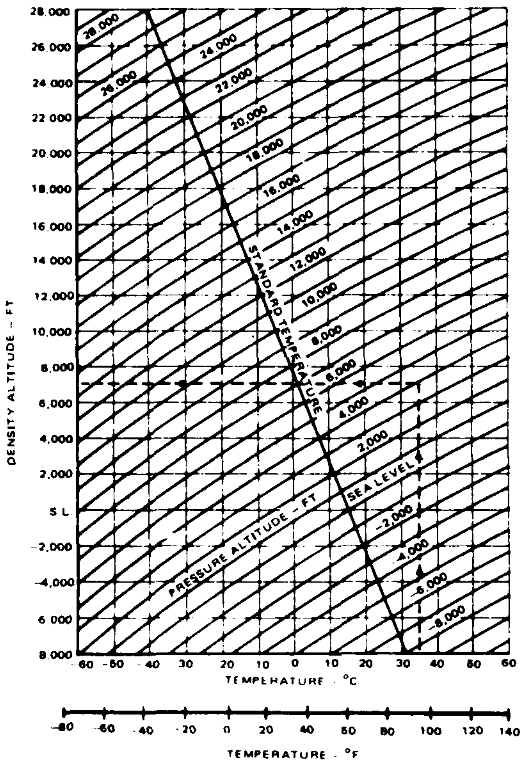

Fig 1 Density Altitude Chart

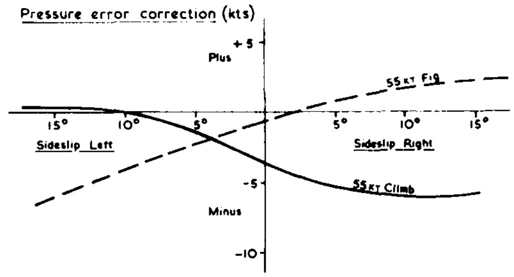

Fig 2 Pressure Error Corrections with Sideslip: Whirlwind Mk10 - XJ 398 (from Ref 8)

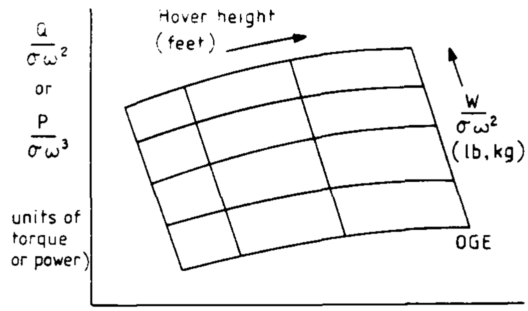

Fig 4 Power Carpet Plot, Hover IGE + OGE (from Ref 8)

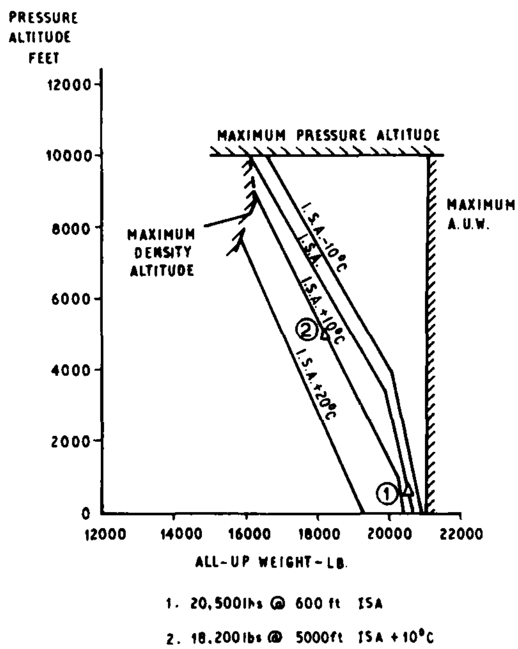

Fig 3 Maximum Weight for Hover OGE (from Ref 8)

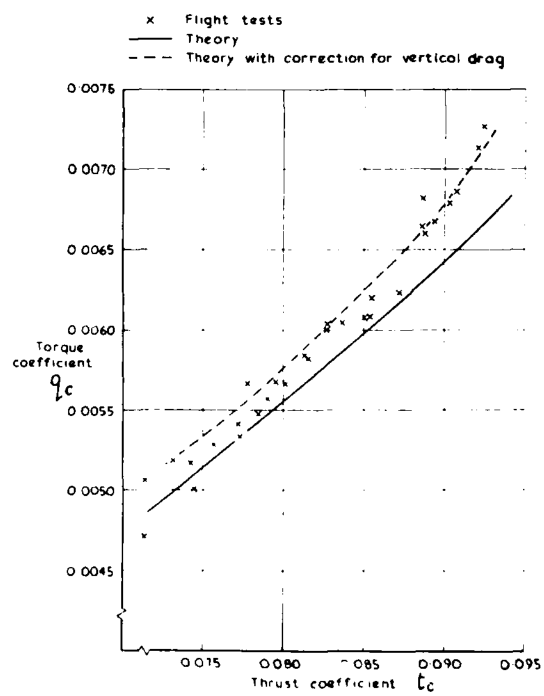

Fig 5 Wessex Main Rotor Thrust and Torque Coefficients (from Ref 20)

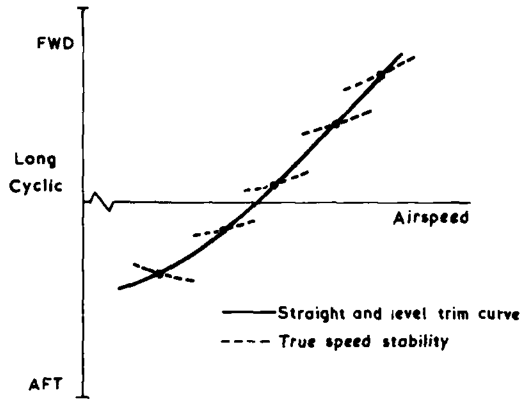

Fig 35 True and Apparent Speed Stability

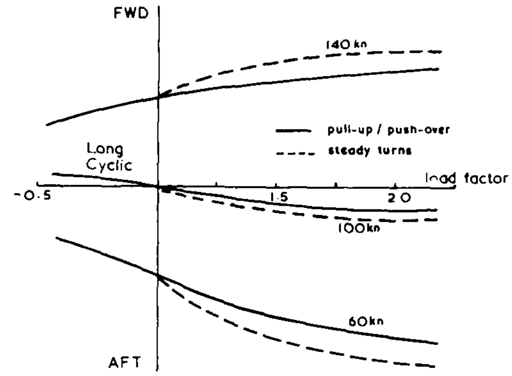

Fig 36 Manoeuvre Stability

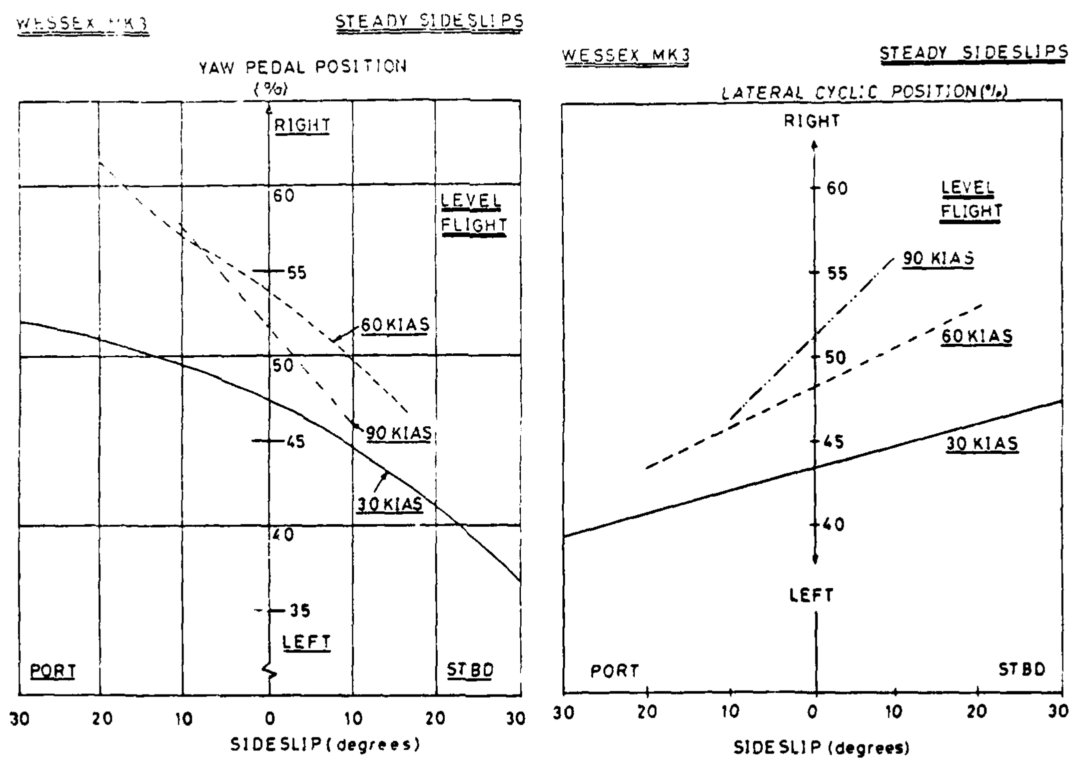

Fig 37 Steady Heading Sideslip Test Results (from Ref 8)

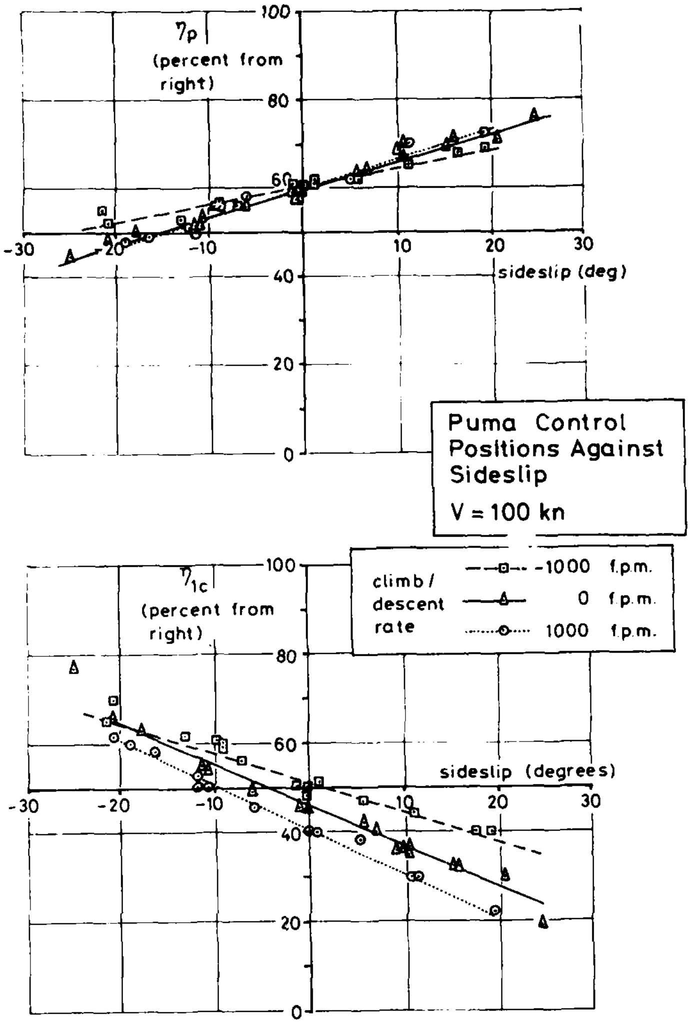

Fig 38 Puma XW 241 - Sideslip Tests

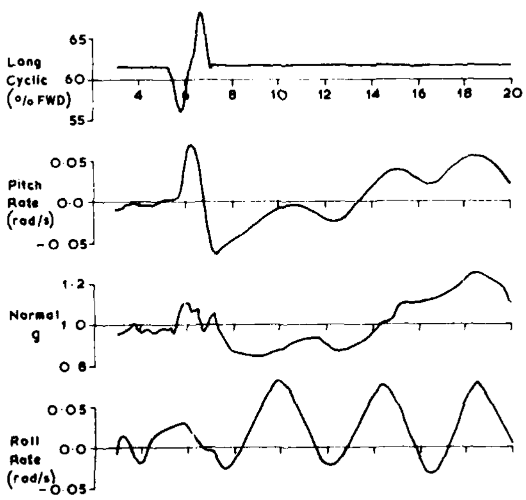

Fig 39 Puma XW 241 - Response to Longitudinal Cyclic Doublet at 100 kn

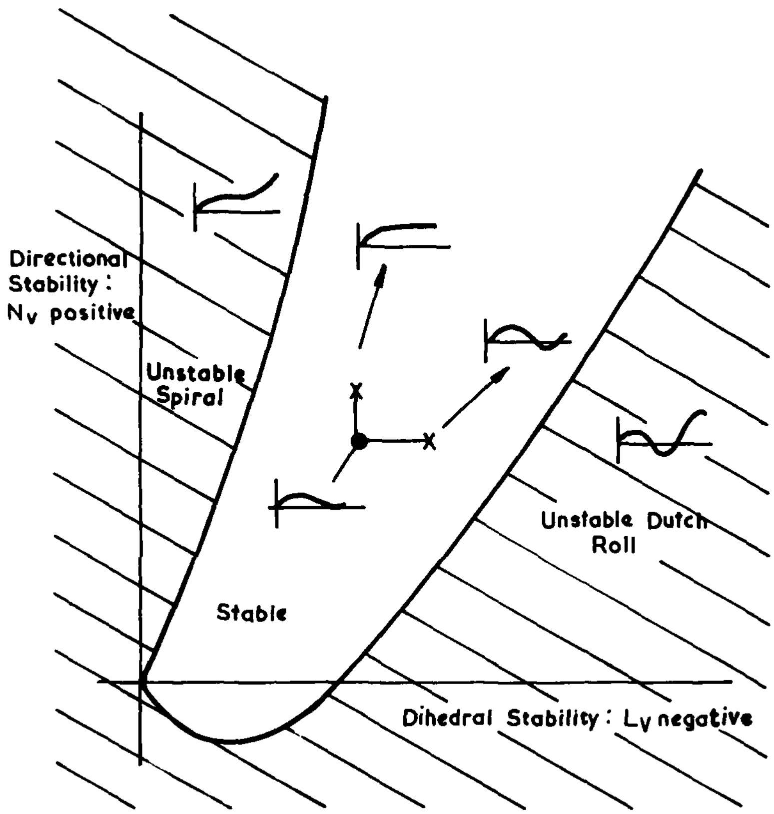

Fig 40 2-Parameter Stability Diagram for Lateral/Directional Motion

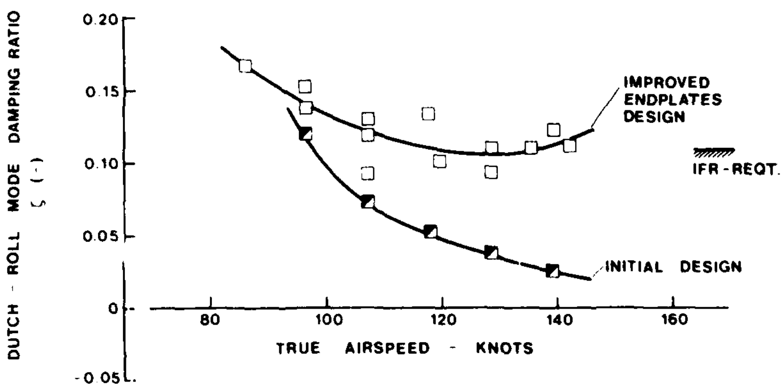

Fig 41 BK 117 Lateral-Directional (Dutch Roll) Mode Damping vs Flight Speed (GW = 2850 kg, Mid CG)

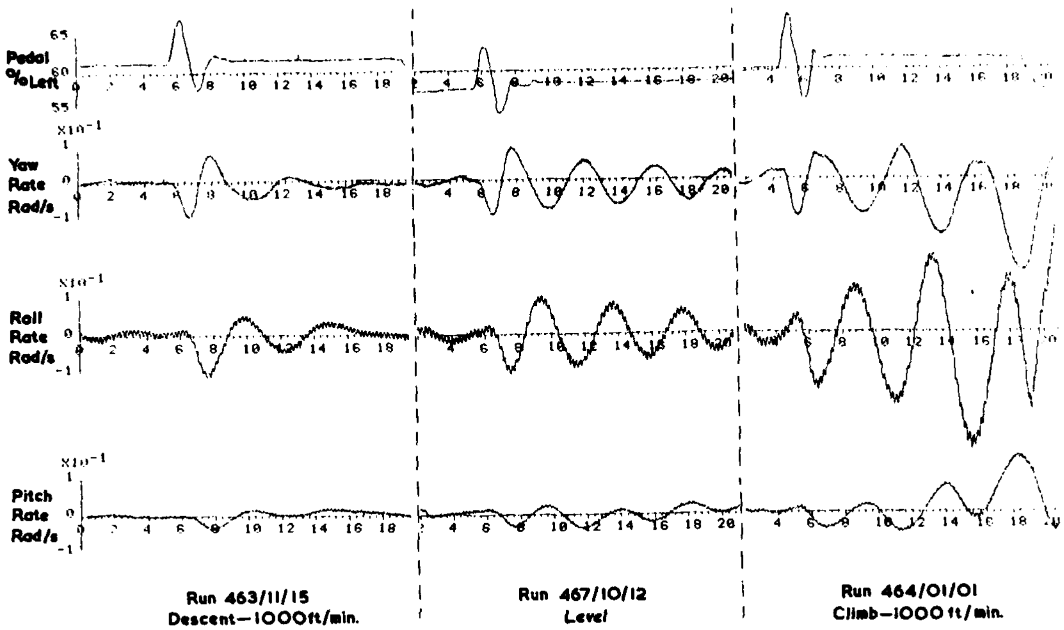

Fig 42 Puma XW 241 - Response to Pedal Doublets at 100 kn

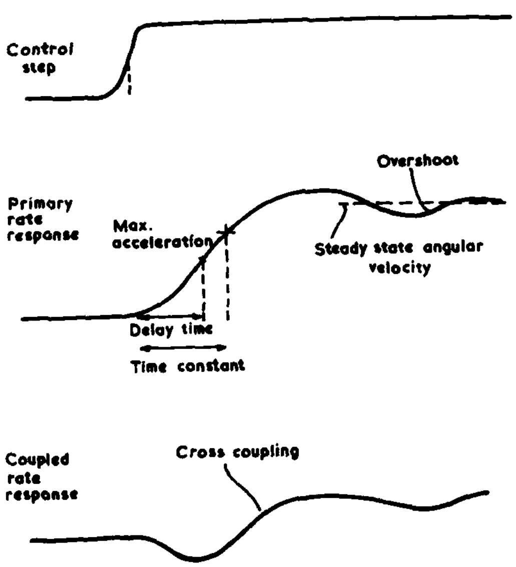

Fig 43 Characteristics of Short Term Control Response

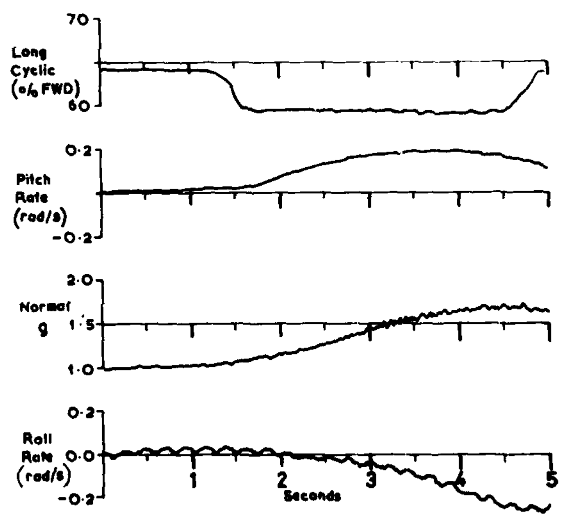

Fig 44 Puma XW 241 - Response to Step Input in Longitudinal Cyclic at 100 kn

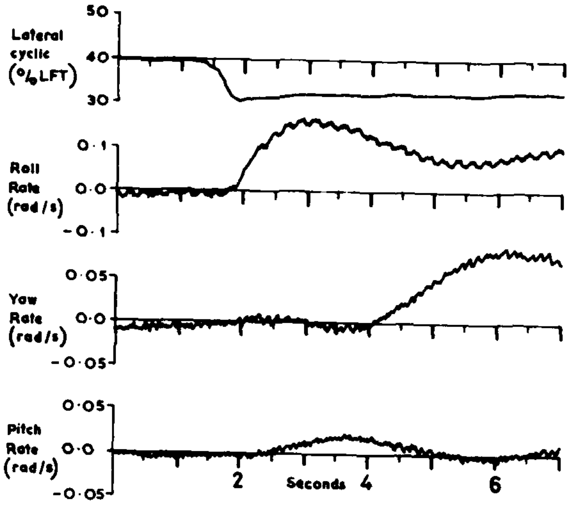

Fig 45 Puma XW 241 - Response to Step Input in Lateral Cyclic at 100 kn

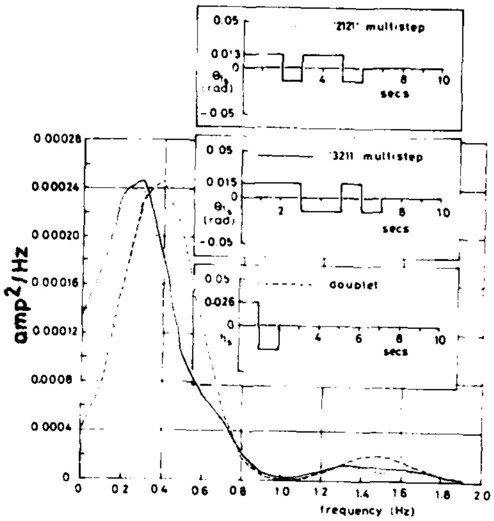

Fig 47 Multi-step Control Inputs and their Spectral Content

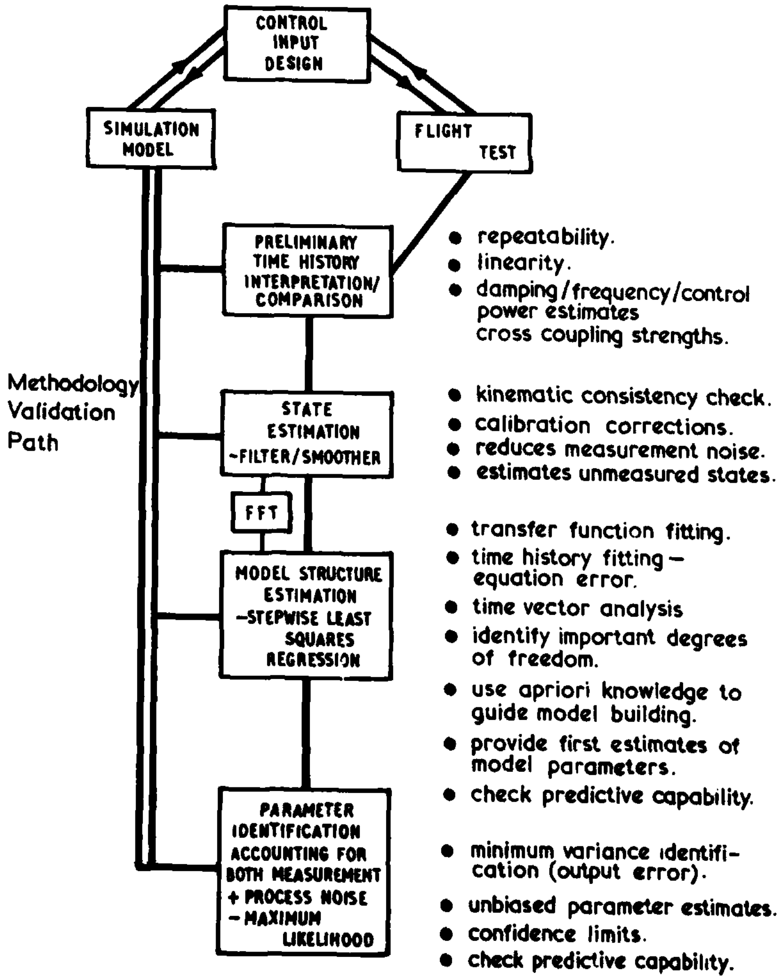

Fig 46 Helicopter Integrated System Identification Methodology

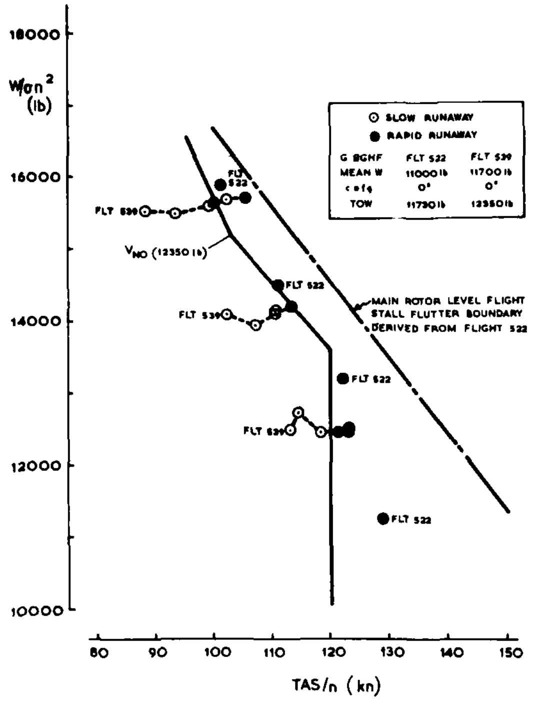

Fig 48 W30 Nose Up Pitch Runaways - Flight Envelope Tests (from Ref 61)

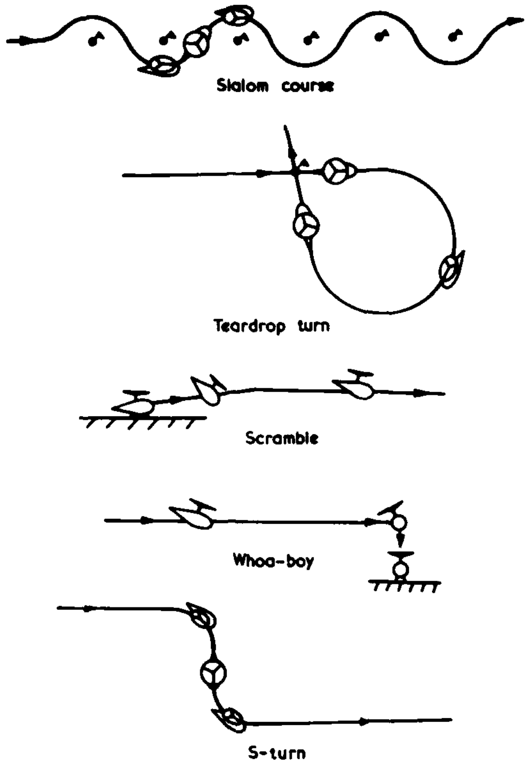

Fig 49 Applied Flying Tasks for NOE Flight (from Ref 62)

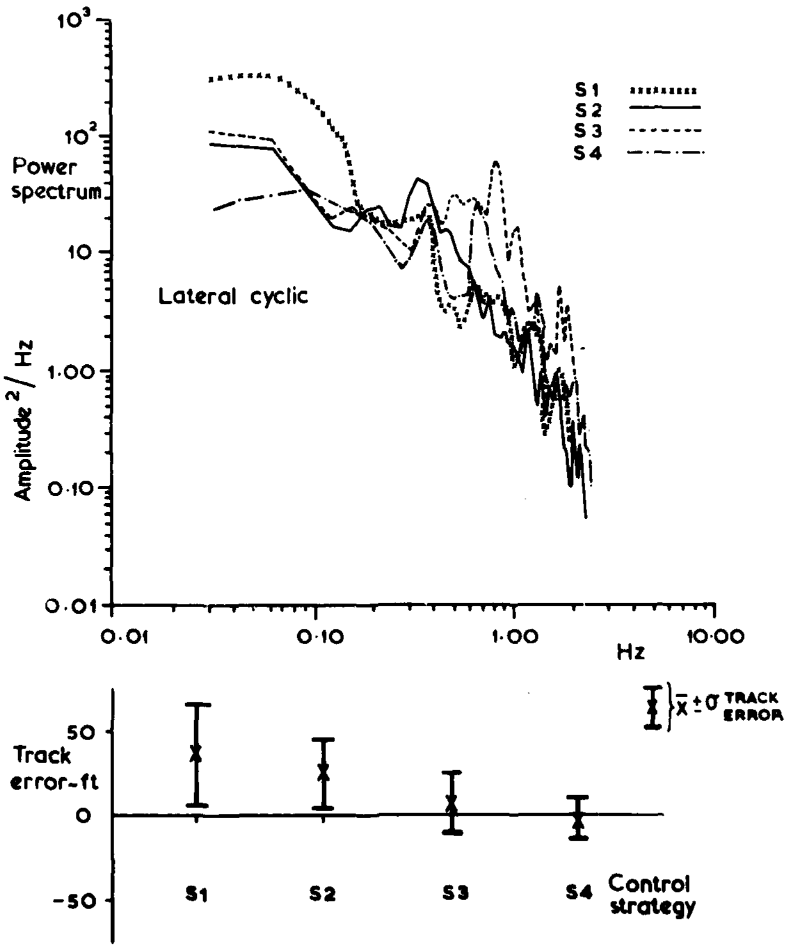

Fig 50 Circular Flight Path Task: Puma Lateral Cyclic Control Power Spectrum and Track Error

BIBLIOGRAPHY

Reference no 1

84A46379 NASA ISSUE 22 Category5

Optimum design of rotor blades for vibration reduction in forward flight

FRIEDMANN, P. P.; SHANTHAKUMARAN, P.

California Univ., Los Angeles. (CD146017)

840000 NSG-1578 IN: American Helicopter Society, Annual Forum, 39th, St. Louis, MO, May 9-11, 1983, Proceedings (A84-46326 22-01). Alexandria, VA, American Helicopter Society, 1984, p. 656-673. Army-supported research. AB(California, University, Los Angeles, CA) 18 p. Refs. 27

Modern structural optimization techniques are applied to vibration reduction of helicopter rotor blades in forward flight. The objective function minimized consists of the oscillatory vertical hub shears or the hub rolling moments at one particular advance ratio. The behavior constraints are the frequency placements of the blade and the requirement that aeroelastic stability margins, in hover, remain unaffected by the optimization process. The aeroelastic stability and response analysis is based on a fully coupled flap-lag-torsional analysis of the blade. Numerical results are presented for some typical soft-in-plane hingeless rotor configurations indicating a 15-40 percent reduction in vibration levels, as well as a blade which is 20 percent lighter than the initial design. These results imply that structural optimization techniques can yield substantial practical benefits in the design process of rotor systems. Author

Controlled Terms: AERODYNAMIC COEFFICIENTS / AERODYNAMIC DRAG / *AEROELASTICITY / *AIRCRAFT STABILITY / *DESIGN ANALYSIS / *HELICOPTERS / INTEGRAL EQUATIONS / OPTIMIZATION / *ROTARY WINGS / *VIBRATION DAMPING

Reference no 2

84A46371 NASA ISSUE 22 Category 5

The utility of speed, agility, and maneuverability for an LHX type mission

VAUSE, R.; HARRIS, M.; FALCO, M.; SHAW, D.; MCDANIEL, R.

840000 IN: American Helicopter Society, Annual Forum, 39th, St. Louis, MO, May 9-11, 1983, Proceedings (A84-46326 22-01). Alexandria, VA, American Helicopter Society, 1984, p. 537-549. AA(Sikorsky Aircraft, Stratford, CT) AB(Flight Systems, Inc., Newport Beach, CA) AC(Grumman Aerospace Corp., Bethpage, NY) AD(System Planning Corp., Arlington, VA) AE(Science and Technology Associates, Inc., Arlington, VA) 13 p. Refs. 7

One of the most exciting and innovative rotary wing programs anticipated in the next decade is the Army's LHX Family of Light Helicopters program. Among the most critical issues yet to be adequately addressed is the utility of speed and agility in the context of the Army's mission in the year 2000. The results of efforts in this area are generalized in ways meaningful to industry and government developers, and to the military requirements writers. The results are presented parametrically in terms such as load factor, speed, acceleration, and deceleration, and measures of effectiveness are used which are applicable to the military environment, e.g. sorties delivered, time on station, response time, and changes in survivability. The results take into account three dimensional considerations, including the effects of both macro and micro terrain. Author

Controlled Terms: ANTIAIRCRAFT MISSILES / COMBAT / *HELICOPTER DESIGN / *LIGHT AIRCRAFT / *MANEUVERABILITY / *MILITARY HELICOPTERS / NAP-OF-THE EARTH NAVIGATION / SYSTEM EFFECTIVENESS / THREAT EVALUATION

Reference no 3

84A46368 NASA ISSUE 22 Category 8

Modelling the effects of blade torsional flexibility in helicopter stability and control

CURTISS, H. C., JR.

840000 IN: American Helicopter Society, Annual Forum, 39th, St. Louis, MO, May 9-11, 1983, Proceedings (A84-46326 22-01). Alexandria, VA, American Helicopter Society, 1984, p. 472-482. Navy-supported research. AA(Princeton University, Princeton, NJ) 11 p. Refs. 13

Various aspects of the coupled flap-torsion motion of an articulated rotor blade are discussed. The rotor blade is assumed to be rigid in the flap-wise direction, with torsional flexibility concetrated at the root of the blade. This analytical model was developed to model the response of a servoflap-controlled rotor. The servoflap rotor is torsionally soft and as a result is very sensitive to aerodynamic modeling assumptions. The influence of modeling assumptions on the flutter boundaries of a rotor blade in hover and the effects of torsional flexibility on flapping behavior of conventional rotors are discussed. It is shown that prediction of the flap-torsion dynamics of torsionally soft rotors is very sensitive to a number of small parameters. The formulation of the aerodynamic forces and moments presented here indicates that torsionally soft rotors have very low levels of damping even for the case of an aerodynamically and inertially balanced rotor blade. Author

Controlled Terms: AERODYNAMIC CHARACTERISTICS / AERODYNAMIC FORCES / *AEROELASTICITY / *AIRCRAFT STABILITY / *CONTROL SIMULATION / FLAPS (CONTROL SURFACES) / FLEXIBILITY / *HELICOPTER CONTROL / PITCHING MOMENTS / ROTARY WINGS / *ROTOR BLADES / SERVOCONTROL / VIBRATION DAMPING

Reference no 4

84A36523 NASA ISSUE 17 Category 53

Flight tests for the assessment of task performance and control activity

PAUSDER, H.-J.; HUMMES, D.

840400 (American Helicopter Society and NASA, Specialists' Meeting on Helicopter Handling Qualities, Moffett Field, CA, April 14, 15, 1982) American Helicopter Society, Journal (ISSN 0002-8711), vol. 29, April 1984, p. 34-41. AB(Deutsche Forschungs- und Versuchsanstalt fuer Luft- und Raumfahrt, Institut fuer Flugmechanik, Brunswick, West Germany) 8 p. Refs. 7

The tests were performed with the helicopters BO 105 and UH-1D. Closely connected with tactical demands the six test pilots' task was to minimize the time and the altitude over the obstacles. The data reduction yields statistical evaluation parameters describing the control activity of the pilots and the achieved task performance. The results are shown in form of evaluation diagrams. Additionally dolphin tests with varied control strategy were performed to get more insight into the influence of control techniques. From these test results

recommendations can be derived to emphasize the direct force control and to reduce the collective to pitch crosscoupling for the dolphin. Previously announced in STAR as N82-23213 T.M.

Controlled Terms: *AIRCRAFT MANEUVERS / ATTITUDE CONTROL / *BO-105 HELICOPTER / CONTROLLABILITY / DATA ACQUISITION / *FLIGHT CONTROL / *FLIGHT TESTS / *HELICOPTER CONTROL / HELICOPTER PERFORMANCE / HOVERING / MILITARY OPERATIONS / *NAP-OF-THE-EARTH NAVIGATION / *PILOT PERFORMANCE / *UH-1 HELICOPTER

Reference no 5

84N33401# NASA ISSUE 23 Category 5
Hover test of a full-scale hingeless rotor
WARMBRODT, W.; PETERSON, R. L.
National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)
NASA-TM-85990; A-9827; NAS 1.15:85990 840800 26 p. Jpn. 3692 HC A03/MF A01

The performance and aeroelastic stability in hover of a 9.8-m diameter, hingeless helicopter rotor system was evaluated. Rotor performance and inplane damping data were obtained for rotor operation between 350 and 425 rpm for thrust coefficients (CT/sigma) between 0.0 and 0.12. At constant rotor thrust, a minimum in rotor inplane damping was measured at 400 rpm. Good agreement is shown between experimental performance data and predicted performance. The influence of different aerodynamic inflow models on predicting damping levels is also shown. The best correlation with experimental stability data was obtained when a dynamic inflow model was used instead of static or quasistatic inflow models. Comparison with other full scale, hingeless rotor data in hover is presented. The hingeless rotor data and data from a full scale, bearingless main rotor test performed on the same general purpose test apparatus were compared. Although the bearingless rotor was more highly damped at design tip speed and 1-g thrust operation, greater sensitivity to operating conditions is shown. At low thrust levels the bearingless main rotor is less damped than the hingeless rotor. E.A.K.

Controlled Terms: AERODYNAMIC CONFIGURATIONS / COMPUTER PROGRAMS / DAMPING TESTS / *GROUND EFFECT (AERODYNAMICS) / GROUND EFFECT MACHINES / *HELICOPTERS / *HOVERING / *RESONANCE / *ROTARY WINGS / *ROTOR AERODYNAMICS

Reference no 6

84A32472 NASA ISSUE 14 Category 8
Lateral-directional stability - Theoretical analysis and flight test experience --- for helicopters
FAULKNER, A.; KLOSTER, M.
MBB-UD-402-83-OE 830900 Associazione Industrie Aerospaziali and Associazione Italiana di Aeronautica ed Astronautica, European Rotorcraft Forum, 9th, Stresa, Italy, Sept. 13-15, 1983. 18 p. AA(Messerschmitt-Boelkow-Blohm GmbH, Munich, West Germany) AB(Muenchen, Fachhochschule, Munich, West Germany) 18 p. Refs. 7

The coupled lateral-directional dynamic stability (Dutch roll) for the helicopter is analysed theoretically using the technique of linearized stability derivatives. Sensitivity studies are used to highlight the most important derivatives for this mode and the model reduction to approximate formulae for the frequency and damping ratio is validated. Data based on parameter identification and a theoretical model are used. The composition of the derivatives is discussed, showing the most important moment and force sources from the rotor, fin and tail rotor. Practical experience from the BK 117 and BO 105 family of helicopters is presented and intepreted. It is shown that nonlinear aerodynamic effects caused by the fuselage and rotor wakes play an important role in the dynamic response and must be considered during the design stage. A balanced tail configuration is suggested. Author

Controlled Terms: AERODYNAMICS / AIRSPEED / BO-105 HELICOPTER / *DIRECTIONAL STABILITY / EIGENVALUES / *FLIGHT STABILITY TESTS / *HELICOPTER PERFORMANCE / *LATERAL STABILITY / PARAMETER IDENTIFICATION / *ROLL / SENSITIVITY / YAW

Reference no 7

84A32471 NASA ISSUE 14 Category 5
Aeromechanical aspects in the design of hingeless/bearingless rotor systems
KLOEPPEL, V.; KAMPA, K.; ISSELHORST, B.
MBB-UD-403-83-OE 830900 Associazione Industrie Aerospaziali and Associazione Italiana di Aeronautica ed Astronautica, European Rotorcraft Forum, 9th, Stresa, Italy, Sept. 13-15, 1983, Paper. 26 p. AC(Messerschmitt-Boelkow-Blohm GmbH, Munich, West Germany) 26 p. Refs. 18

Theoretical models are used in a discussion of the pronounced torsion-flap-lag coupling shown by hingeless/bearingless rotors which significantly influences general aeromechanical characteristics of the helicopter. The analysis uses an equivalent system technique, where the position of the blade relative to the control axis is determined with the help of the blade-to-beam angle, Beta-BB. Because the built-in value of this parameter may change elastically depending on flight condition, the actual size of the effective Beta-BB has to be estimated beforehand using a specialized beam theory or finite element method. In the design stage, Beta-BB and the control stiffness provide pitch-lag coupling which can be used to increase the lead-lag damping in the rotating and in the fixed system for ground and air resonance conditions. Theoretical results are verified by whirl tower and flight test data for the BO105 hingeless rotor and the bearingless tail rotor on BK117. J.N.

Controlled Terms: *AEROELASTICITY / *BEARINGLESS ROTORS / COUPLING / DAMPING / *HELICOPTER DESIGN / PITCH (INCLINATION) / RESONANCE TESTING / *RIGID ROTORS / STIFFNESS / STRUCTURAL STABILITY

Reference no 8

84A31713 NASA ISSUE 13 Category 5
Unsteady aerodynamics in time and frequency domains for finite time arbitrary motion of rotary wings in hover and forward flight
DINYAVARI, M. A. H.; FRIEDMANN, P. P.
California Univ., Los Angeles. (CD146017)
AIAA PAPER 84-0988 840000 NAG2-209 IN: Structures, Structural Dynamics and Materials Conference, 25th, Palm Springs, CA, May 14-16, 1984, and AIAA Dynamics Specialists Conference, Palm Springs, CA, May 17, 18, 1984, Technical Papers. Part 2 (A84-31684 13-39). New York, American Institute of Aeronautics and Astronautics, 1984, p. 266-282. AB(California, University, Los Angeles, CA) 17 p. Refs. 30 Jpn. 1835

Several incompressible finite-time arbitrary-motion airfoil theories suitable for coupled flap-lag-torsional aeroelastic analysis of helicopter rotors in hover and forward flight are derived. These theories include generalized Greenberg's theory, generalized Loewy's theory, and a staggered cascade theory. The generalized Greenberg's and staggered cascade theories were derived directly in Laplace domain considering the finite length of the wake and using operational methods. The load expressions are presented in Laplace, frequency, and time domains. Approximate time domain loads for the various generalized theories, discussed in the paper, are obtained by developing finite state models using the Pade approximant of the appropriate lift deficiency functions. Three different methods for constructing Pade approximants of the lift deficiency functions were considered and the more flexible one was used. Pade approximants of Loewy's lift deficiency function, for various wake spacing and radial location parameters of a helicopter typical rotor blade section, are presented. Author

Controlled Terms: AERODYNAMIC LOADS / *AERODYNAMIC STABILITY / *AEROELASTICITY / AIRFOILS / HELICOPTERS / *HOVERING STABILITY / PADE APPROXIMATION / *ROTARY STABILITY / *ROTARY WINGS / TORSION / TWO DIMENSIONAL FLOW / WAKES

Reference no 9

84N29869# NASA ISSUE 20 Category 5

Improved pilot model for application to a computer flight testing program for helicopters

HAVERDINGS, H.

National Aerospace Lab., Amsterdam (Netherlands). (NE736790)

NLR-MP-83052-U 830826 Flight Div. Sponsored by Royal Netherlands Air Force Directorate of Material Air Presented at 9th European Rotorcraft Forum, Stress, Italy, 13-15 Sep. 1983 31 p. Jpn. 3129 HC A03/MF A01

A computer flight testing program for evaluation of helicopter dynamics and handling and control qualities was developed. A control model to fly the mathematical helicopter model was designed for maneuvering flight. The model was refined to improve controllability so as to be able to keep state variable deviations small during a maneuver. This required augmentation of the cost function (to be optimized) by specifying weighting terms on control rate and on the control and the state as well. Terminal conditions have to be met approximately, rather than exactly, so as to eliminate the problem of the pilot's feedback gains becoming infinitely large. Results of an application for a landing flare with an S-61 class helicopter are shown. Author (ESA)

Controlled Terms: *AIRCRAFT MANEUVERS / COMPUTER SYSTEMS PROGRAMS / *COMPUTERIZED SIMULATION / CONTROLLABILITY / *FLIGHT SIMULATION / *HELICOPTER CONTROL / MAN MACHINE SYSTEMS / *MATHEMATICAL MODELS / OPTIMAL CONTROL / PILOT PERFORMANCE / S-61 HELICOPTER / TURNING FLIGHT

Reference no 10

84N28806# NASA ISSUE 19 Category 8

UH-60A Black Hawk engineering simulation program. Volume 1: Mathematical model

Final Report

HOWLETT, J. J.

Sikorsky Aircraft, Stratford, Conn. (SL280853)

NASA-CR-166309; NAS 1.26:166309; SER-70452 811200 NAS2-10626 359 p. Jpn. 2968 HC A16/MF A01

A nonlinear mathematical model of the UR-60A Black Hawk helicopter was developed. This mathematical model, which was based on the Sikorsky General Helicopter (Gen Hel) Flight Dynamics Simulation, provides NASA with an engineering simulation for performance and handling qualities evaluations. This mathematical model is total systems definition of the Black Hawk helicopter represented at a uniform level of sophistication considered necessary for handling qualities evaluations. The model is a total force, large angle representation in six rigid body degrees of freedom. Rotor blade flapping, lagging, and hub rotational degrees of freedom are also represented. In addition to the basic helicopter modules, supportive modules were defined for the landing interface, power unit, ground effects, and gust penetration. Information defining the cockpit environment relevant to pilot in the loop simulation is presented. Author

Controlled Terms: AIRCRAFT LANDING / *COCKPITS / *CONTROLLABILITY / DOCUMENTATION / ENGINE CONTROL / FUSELAGES / GUST LOADS / *H-60 HELICOPTER / *HELICOPTER CONTROL / *HELICOPTER DESIGN / HELICOPTER ENGINES / HELICOPTER TAIL ROTORS / *MATHEMATICAL MODELS / ROTARY WINGS / SIMULATION / SYSTEMS ANALYSIS

Reference no 11

84N37695# NASA ISSUE 18 Category 2

The investigation of a variable camber blade lift control for helicopter rotor systems

AWANI, A. O.

Kansas Univ. Center for Research, Inc., Lawrence. (KF728369)

NASA-CR-3505; NAS 1.26:3505 820100 NCC-292 130 p. Jpn. 2788 HC A07/MF A01

A new rotor configuration called the variable camber rotor was investigated numerically for its potential to reduce helicopter control loads and improve hover performance. This rotor differs from a conventional rotor in that it incorporates a deflectable 50% chord trailing edge flap to control rotor lift, and a non-feathering (fixed) forward portion. Lift control is achieved by linking the blade flap to a conventional swashplate mechanism; therefore, it is pilot action to the flap deflection that controls rotor lift and tip path plane tilt. This report presents the aerodynamic characteristics of the flapped and unflapped airfoils, evaluations of aerodynamics techniques to minimize flap hinge moment, comparative hover rotor performance and the physical concepts of the blade motion and rotor control. All the results presented herein are based on numerical analyses. The assessment of payoff for the total configuration in comparison with a conventional blade, having the same physical characteristics as an H-34 helicopter rotor blade was examined for hover only. B.W.

Controlled Terms: *AIRFOILS / CAMBER / COMPUTER PROGRAMS / *HELICOPTER TAIL ROTORS / *HELICOPTERS / *HOVERING / LIFT / *LOADS (FORCES) / MATHEMATICAL MODELS / PITCHING MOMENTS / PRESSURE DISTRIBUTION / TRAILING-EDGE FLAPS

Reference no 12

84A25553 NASA ISSUE 10 Category 8

Models and analysis for twin-lift helicopter systems

LEWIS, J.; MARTIN, C.

National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)

830000 NAG2-82; NSG-2384; DE-AC01-80RA-50256 IN: American Control Conference, San Francisco, CA, June 22-24, 1983, Proceedings. Volume 3 (A84-25451 10-63). New York, Institute of Electrical and Electronics Engineers, 1983, p. 1324, 1325. AA(NASA, Ames Research Center, Moffett Field, CA) AB(Case Western Reserve University, Cleveland, OH) 2 p. Refs. 6

The major limitation of a modern helicopter with respect to its employment in transport operations has been a relatively small payload size. Among several ideas for overcoming this limitation are a number of multiple-lift schemes using two or more helicopters to lift payloads too large for a single helicopter. On a number of occasions, pairs of helicopters carrying a single load have been successfully employed. However, a considerable amount of flying skill was required in these cases in connection with the manual control tasks involved. Some form of automatic control will, therefore, be needed until twin-lift operations become generally practical. The present investigation is concerned with the development of models of the Twin-Lift System which can be used as a basis for the design of automatic control. G.R.

Controlled Terms: *AIRCRAFT CONTROL / *AUTOMATIC CONTROL / *CONTROL THEORY / DESIGN ANALYSIS / DIFFERENTIAL EQUATIONS / *HEAVY LIFT HELICOPTERS / *HELICOPTER DESIGN / MATHEMATICAL MODELS / MATRICES (MATHEMATICS) / PAYLOADS

Reference no 13

84N20571# NASA ISSUE 11 Category 8

A summary of rotorcraft handling qualities research at NASA Ames Research Center

CHEN, R. T. N.

National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)

840300 In NASA. Langley Research Center NASA Aircraft Controls Research, 1983 p 51-68 (SEE N84-20567 11-08) 18 p. Jpn. 1615 HC A25/MF A01

The objectives of the rotorcraft handling qualities research program at Ames Research Center are twofold: (1) to develop basic handling qualities design criteria to permit cost effective design decisions to be made for helicopters, and (2) to obtain basic handling qualities data for certification of new rotorcraft configurations. The research on the helicopter handling qualities criteria has focused primarily on military nap-of-the-earth (NOE) terrain flying missions, which are flown in day visual meteorological conditions (VMC) and instrument meteorological conditions (IMC), or at night. The Army has recently placed a great deal of emphasis on terrain flying tactics in order to survive and effectively complete the missions in modern and future combat environments. Unfortunately, the existing Military Specification MIL-H 8501A which is a 1961 update of a 1951 document, does not address the handling qualities requirements for terrain flying. The research effort is therefore aimed at filling the void and is being conducted jointly with the Army Aeromechanics Laboratory at Ames. The research on rotorcraft airworthiness standards with respect to flying qualities requirements was conducted to collaboration with the Federal Aviation Administration (FAA). Author

Controlled Terms: AERODYNAMIC STABILITY / *AUGMENTATION / *CONTROLLABILITY / COST EFFECTIVENESS / DESIGN ANALYSIS / *HELICOPTERS / LOADS (FORCES) / *NAP-OF-THE-EARTH NAVIGATION / THRUST CONTROL / VERTICAL MOTION SIMULATORS

Reference no 14

84A19744 NASA ISSUE 7 Category 3

Simulator investigations of side-stick controller/stability and control augmentation systems for night nap-of-earth flight

LANDIS, K. H.; AIKEN, E. W.

Boeing Vertol Co., Philadelphia, Pa. (BR870123)

840100 NAS2-10880 American Helicopter Society, Journal (ISSN 0002-8711), vol. 29, Jan. 1984, p. 56-65. AA(Boeing Vertol Co., Philadelphia, PA) AB(U.S. Army, Aeromechanics Laboratory, Moffett Field, CA) 10 p. Refs. 5 Jpn. 865

Several night nap-of-the-earth mission tasks were evaluated using a helmet-mounted display which provided a limited field-of-view image with superimposed flight control symbology. A wide range of stability and control augmentation designs was investigated. Variations in controller force-deflection characteristics and the number of axes controlled through an integrated side-stick controller were studied. In general, a small displacement controller is preferred over a stiffstick controller particularly for maneuvering flight. Higher levels of stability augmentation were required for IMC tasks to provide handling qualities comparable to those achieved for the same tasks conducted under simulated visual flight conditions. Previously announced in STAR as N82-23216 T.M.

Controlled Terms: AIRCRAFT MANEUVERS / *CONTROLLABILITY / *CONTROLLERS / *FLIGHT CONTROL / FLIGHT SIMULATION / *HELICOPTER CONTROL / HELMET MOUNTED DISPLAYS / MILITARY HELICOPTERS / *NAP-OF-THE-EARTH NAVIGATION / *NIGHT FLIGHTS (AIRCRAFT) / NIGHT VISION / PILOT PERFORMANCE / *STABILITY AUGMENTATION / VISUAL FLIGHT

Reference no 15

84A19657 NASA ISSUE 7 Category 2

Advanced rotor analysis methods for the aerodynamics of vortex/blade interactions in hover

SUMMA, J. M.

820800 DAAK51-81-C-0006; DAAG29-81-C-0032 Association Aeronautique et Astronautique de France, European Rotorcraft Forum, 8th, Aix-en-Provence, France, Aug. 31-Sept. 3, 1982, Paper. 18 p. AA(Analytical Methods, Inc., Bellevue, WA) 18 p. Refs. 17 Jpn. 860

The work discussed in this report has shown that the complete hovering rotor wake geometry, including the inner sheet, can be predicted without the constraints or empiricisms of a prescribed wake. Moreover, the calculated wakes for some modern rotors violate the usual hypothesis in prescribed wake methods of a weak linear inner sheet and a single rolled-up tip vortex. When coupled with a lifting-surface method, this relaxed wake procedure allows for the accurate analysis of rotor performance at proper collective settings.

Finally, the application of a surface singularity method developed for rotors has demonstrated the capability of accurately computing blade surface pressures very near the rotor tip edge. Author

Controlled Terms: *COMPUTATIONAL FLUID DYNAMICS / COMPUTER PROGRAMS / *HELICOPTER WAKES / *HOVERING / *INTERACTIONAL AERODYNAMICS / PRESSURE DISTRIBUTION / *ROTOR AERODYNAMICS / ROTOR BLADES / *VORTEX SHEDDING / WING TIP VORTICES

Reference no 16

84A19644 NASA ISSUE 7 Category 8

Techniques in the assessment of helicopter flying qualities

OBERMAYER, M.; FAULKNER, A.

820800 Association Aeronautique et Astronautique de France, European Rotorcraft Forum, 8th, Aix-en-Provence, France, Aug. 31-Sept. 3, 1982, Paper. 22 p. AB(Messerschmitt-Boelkow-Blohm GmbH, Munich, West Germany) 22 p. Refs. 14 Jpn. 872

The reasons why safety, control precision, and passenger comfort are of primary importance in the assessment of helicopter flying qualities are discussed. The benefit of correlating the handling qualities to actual helicopter properties by simulation studies is stressed. A more restricted range of control power and damping is recommended, contrary to a NASA study. Studies for cross-axis coupling and for pitch and roll cross-coupling do not concur. A pilot questionnaire, parameter identification and frequency response methods are discussed as means of test flight interpretation, facilitated by pulse code manipulation and digital data processing. C.M.

Controlled Terms: *CONTROLLABILITY / *CROSS COUPLING / DYNAMIC STABILITY / FLAPPING / *FLIGHT CHARACTERISTICS / FLIGHT TESTS / *HELICOPTER CONTROL / *HELICOPTER PERFORMANCE / INSTRUMENT FLIGHT RULES / LATERAL CONTROL / LONGITUDINAL CONTROL / PARAMETER IDENTIFICATION / SAFETY FACTORS / SIMULATION

Reference no 17

84A19603 NASA ISSUE 7 Category 8

A simple system for helicopter Individual-Blade-Control and its application to lag damping augmentation

HAM, N. D.; BEHAL, B. L.; MCKILLIP, R. M., JR.

Massachusetts Inst. of Tech., Cambridge. (MJ700802)

820800 NSG-2266 Association Aeronautique et Astronautique de France, European Rotorcraft Forum, 8th, Aix-en-Provence, France, Aug. 31-Sept. 3, 1982, Paper. 10 p. NASA-sponsored research. AC(MIT, Cambridge, MA) 10 p. Refs. 8 Jpn. 872

A new, advanced type of active control for helicopters and its application to a system for blade lag damping augmentation is described. The system, based on previously developed M.I.T. Individual-Blade-Control hardware, employs blade-mounted accelerometers to sense blade lag motion and feeds back rate information to increase the damping of the first lag mode. A linear model of the blade and control system dynamics is used to give guidance in the design process as well as to aid in analysis of experimental results. System performance in wind tunnel tests is described, and evidence is given of the system's ability to provide substantial additional damping to blade lag motion. Author

Controlled Terms: CORIOLIS EFFECT / *HELICOPTER CONTROL / *ROTARY STABILITY / *ROTARY WINGS / *ROTOR AERODYNAMICS / VIBRATION DAMPING / WIND TUNNEL TESTS

Reference no 18

84A19601 NASA ISSUE 7 Category 8

The effects of pilot stress factors on handling quality assessments during US/German helicopter agility flight tests

PAUSDER, H.-J.; GERDES, R. M.

Institut fuer Flugmechanik, Brunswick (West Germany). (IH545557)

820800 Association Aeronautique et Astronautique de France, European Rotorcraft Forum, 8th, Aix-en-Provence, France, Aug. 31-Sept. 3, 1982, Paper. 19 p. AA(Deutsche Forschungs- und Versuchsanstalt fuer Luft- und Raumfahrt, Institut fuer Flugmechanik, Brunswick, West Germany) AB(NASA, Ames Research Center, Moffett Field, CA) 19 p. Refs. 14 Jpn. 872

Flight tests were conducted with two helicopters to study and evaluate the effects of helicopter characteristics and pilot and task demands on performance in nap-of-the-earth flight. Different, low-level slalom courses were set up and were flown by three pilots with different levels of flight experience. A pilot rating questionnaire was used to obtain redundant information and to gain more insight into factors that influence pilot ratings. The flight test setups and procedures are described, and the pilot ratings are summarized and interpreted in close connection with the analyzed test data. Pilot stress is discussed. The influence of demands on the pilot, of the helicopter characteristics, and of other stress factors are outlined with particular emphasis on how these factors affect handling-qualities assessment. Previously announced in STAR as N83-13114 Author

Controlled Terms: *CONTROLLABILITY / DATA ACQUISITION / *FLIGHT STRESS (BIOLOGY) / *FLIGHT TESTS / *HELICOPTER PERFORMANCE / INTERNATIONAL COOPERATION / NASA PROGRAMS / *PILOT PERFORMANCE / STRESS (PSYCHOLOGY) / TASK COMPLEXITY

Reference no 19

84N19296# NASA ISSUE 10 Category 2

A time domain analysis of a rigid two-bladed fully gimballed helicopter rotor with circulation control

Final Report

MONTANA, P. S.

Naval Ship Research and Development Center, Bethesda, Md. (NT691395)

AD-A136947; AD-E001638; DTNSRDC-83/081; AERO-1282 831200 ZRO-2302 88 p. Jpn. 1422 HC A05/MF A01

An analytic investigation was made to determine the dynamic properties of a two-bladed rigid fully gimballed helicopter rotor incorporating circulation control airfoils and tip jet propulsion. A time domain analysis was developed which provided the capability of using nonlinear airfoil aerodynamics and arbitrary rotor physical characteristics. The effects of feather principal axis of inertia location, horizontal gust disturbances, and feedback control on rotor stability were assessed. Results of the investigation indicate that the subject helicopter rotors are unstable in forward flight without feedback control. With feedback, the rotors are stable and controllable. Author (GRA)

Controlled Terms: *AERODYNAMIC STABILITY / AIRFOILS / BLADE TIPS / *CIRCULATION CONTROL ROTORS / *COMPUTERIZED SIMULATION / CRUISING FLIGHT / EQUATIONS OF MOTION / *GIMBALS / *GUSTS / *HEAVY LIFT HELICOPTERS / HORIZONTAL ORIENTATION / JET PROPULSION / LIFT / *RIGID ROTOR HELICOPTERS / SLOTS / STABILIZERS (FLUID DYNAMICS) / TIME SERIES ANALYSIS

Reference no 20

84N16215# NASA ISSUE 7 Category 8

A helicopter flight investigation of roll-control sensitivity, damping and cross coupling in a low altitude lateral maneuvering task

CORLISS, L. D.; CARICO, D.

National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)

NASA-TM-84376; A-9431; NAS 1.15:84376 831200 Sponsored in part by Army Aviation Research and Development Command 50 p. Jpn. 959 HC A03/MF A01

A helicopter in-flight simulation was conducted to determine the effects of variations in roll damping, roll sensitivity, and pitch and roll rate cross-coupling on helicopter flying qualities in a low altitude maneuver. The experiment utilized the UH-1H helicopter in-flight simulator, which is equipped with the V/STOLAND avionics system. The response envelope of this vehicle allowed simulation of configurations with low to moderate damping and sensitivity. A visual, low level slalom course was set up, consisting of constant speed and constant altitude S-turns around the 1000 ft makers of an 8000 ft runway. Results are shown in terms of Cooper-Harper pilot ratings, pilot commentary, and statistical and frequency analyses of the lateral characteristics. These results show good consistency with previous ground simulator results and are compared with existing flying qualities criteria. S.L.

Controlled Terms: *AIRCRAFT MANEUVERS / AVIONICS / CROSS COUPLING / *FLIGHT CHARACTERISTICS / FLIGHT SIMULATION / *HELICOPTERS / *LATERAL CONTROL / *LOW ALTITUDE / NAP-OF-THE-EARTH NAVIGATION / *PITCH (INCLINATION) / SENSITIVITY

Reference no 21

84N16173# NASA ISSUE 7 Category 5

Helicopter flying qualities characteristics-CH-46E, volume 4

Final Report

Boeing Vertol Co., Philadelphia, Pa. (BR870123)

AD-A134323; NADC-81118-60-VOL-4 831003 524 p. Warminster, Pa. Naval Air Development Center Jpn. 951 HC A22/MF A01

For complete abstract, See AD-A134320. This volume contains the trim and stability derivative output data obtained from the Boeing Vertol Tandem Rotor Trim and Stability Analysis Program (A-97), for the CH-46E helicopter. GRA

Controlled Terms: AERODYNAMIC BALANCE / AERODYNAMIC CHARACTERISTICS / *AERODYNAMIC STABILITY / *FLIGHT CHARACTERISTICS / FLIGHT CONTROL / *HELICOPTER PERFORMANCE / *HELICOPTERS / ROTARY WINGS / STABILITY AUGMENTATION / STABILITY DERIVATIVES

Reference no 22

84N16170# NASA ISSUE 7 Category 5

Helicopter flying qualities characteristics-CH-46E, volume 1

Final Report

Boeing Vertol Co., Philadelphia, Pa. (BR870123)

AD-A134320; NADC-81118-60-VOL-1 831003 160 p. Warminster, Pa. Naval Air Development Center Jpn. 951 HC A08/MF A01

This document defines the flying qualities characteristics of the CH-46E helicopter. The data are representative of both the metal-bladed and composite-bladed versions. Analytically computed static trim data are presented for a wide range of configurations (gross weight, c.g.) and flight conditions (airspeed, altitude, sideslip, climb, autorotation). Correlation of trim data with available flight test data is provided for validation. Analytically computed static stability and control derivatives are compiled for significant combinations of configuration and flight condition. Time history data relating to dynamic stability, control response and Stability Augmentation System failures are extracted from flight test records obtained during the contractor's CH-46E SLEP 2 flight test program. GRA

Controlled Terms: AERODYNAMIC BALANCE / AERODYNAMIC CHARACTERISTICS / *AERODYNAMIC STABILITY / AIRCRAFT MANEUVERS / COMPOSITE STRUCTURES / COMPUTERIZED SIMULATION / DRAG COEFFICIENTS / *FLIGHT CHARACTERISTICS / FLIGHT CONTROL / FLIGHT TESTS / *HELICOPTER PERFORMANCE / *HELICOPTERS / LIFT AUGMENTATION / ROTARY WINGS / SENSITIVITY / STABILITY AUGMENTATION / STATIC STABILITY

Reference no 23

84A16159 NASA ISSUE 5 Category 5

The challenges of maneuvering flight performance testing in modern rotary wing aircraft

PARLIER, C. A.

830000 IN: 1983 report to the aerospace profession; Proceedings of the Twenty-seventh Symposium, Beverly Hills, CA, September 28-October 1, 1983 (A84-16157 05-05). Lancaster, CA, Society of Experimental Test Pilots, 1983, p. 21-41. AA(Hughes Helicopters, Inc., Culver City, CA) 21 p.

The increasing demand for greater maneuvering flight performance in rotary wing aircraft has illuminated a weakness which faces each segment of the rotary wing aviation community. Helicopter specifications are inadequate in their definition of maneuvering performance. The actual maneuvering performance of modern rotary wing aircraft can be greater than the demonstrated maneuvering performance required by the specification. Pilots are given a flight envelope which they can not identify due to insufficient presentation of maneuvering flight information. The paper presents several broad areas where this weakness is manifested. Specific improvements incorporated in the U.S. Army/Hughes Helicopters, Inc., AH-64A Apache are discussed. Author

Controlled Terms: *AIRCRAFT MANEUVERS / COCKPITS / *FLIGHT TESTS / *HELICOPTER PERFORMANCE / MILITARY HELICOPTERS / OPERATIONAL PROBLEMS / *ROTARY WINGS / TERRAIN FOLLOWING AIRCRAFT

Reference no 24

84A15997 NASA ISSUE 5 Category 8

Flight testing the Rotor Systems Research Aircraft (RSRA)

MERRILL, R. K.; HALL, G. W.

Army Aviation Engineering Flight Activity, Edwards AFB, Calif. (AZ102498)

820000 IN: 1982 report to the aerospace profession; Proceedings of the Twenty-sixth Symposium, Beverly Hills, CA, September 22-25, 1982 (A84-15976 05-05). Lancaster, CA, Society of Experimental Test Pilots, 1982, p. 363-373. AA(U.S. Army, Aviation Engineering Flight Activity, Edwards AFB, CA) AB(NASA, Ames Research Center, Moffett Field, CA) 11 p.

The Rotor Systems Research Aircraft (RSRA) is a dedicated rotor test vehicle whose function is to fill the gap between theory, wind tunnel tests and flight verification data. Its flight test envelope has been designed to encompass the expected envelopes of future rotor systems under all flight conditions. The test configurations of the RSRA include pure helicopter and compound (winged helicopter) modes. In addition, should it become necessary to jettison an unstable rotor system in flight, the RSRA may be flown as a fixed wing aircraft. The heart of the RSRA's electronic flight control system is the TDY-43 computer, which can be programmed in numerous ways to change stability and control or force feel system gains. Computer programming changes allow the RSRA to be used as a five-degree-of-freedom inflight simulator for studying the handling qualities of research rotors. O.C.

Controlled Terms: AIRBORNE/SPACEBORNE COMPUTERS / *AIRCRAFT CONTROL / CONTROLLABILITY / DEGREES OF FREEDOM / *ELECTRONIC CONTROL / *FLIGHT CONTROL / *FLIGHT TESTS / HELICOPTER CONTROL / *ROTOR SYSTEMS RESEARCH AIRCRAFT / *TEST VEHICLES / WINGED VEHICLES

Reference no 25

84A12331 NASA ISSUE 2 Category 5

The challenges of maneuvering flight performance testing in modern rotary wing aircraft

PARLIER, C. A.

AIAA PAPER 83-2739 83110C AIAA, AHS, IES, SETP, SFTE, and DGLR, Flight Testing Conference. 2nd, Las Vegas, NV, Nov. 16-18, 1983. 10 p. AA(Hughes Helicopters, Inc., Mesa, AZ) 10 p.

The present investigation is concerned with the maneuvering capability of current and future rotary wing aircraft as one of the important aspects of flight testing. For the objectives of the investigation, maneuverability is defined as the true limit performance of an aircarft. Background developments related to increased needs for greater maneuverability in helicopters are examined, taking into account the first general use of helicopters to perform ground attack missions in the mid-sixties, and the advent of potential air-to-air weaponry on the Mi-24E Hind. Attention is given to evasive maneuvering (EVM), needs for greater maenuverability in connection with low level terrain flight, specification limiations, rotary wing maneuvering flight shortcomings, and proposed improvements. G.R.

Controlled Terms: AH-64 HELICOPTER / *AIRCRAFT MANEUVERS / AIRCRAFT SPECIFICATIONS / *FLIGHT TESTS / *HELICOPTER PERFORMANCE / MILITARY HELICOPTERS / *ROTARY WING AIRCRAFT / TERRAIN FOLLOWING AIRCRAFT

Reference no 26

84A12328 NASA ISSUE 2 Category 5

Flight test technique for the assessment of helicopter mission demands

PAUSDER, H.-J.; MEYER, H.; SANDERS, K.; WULFF, G.

AIAA PAPER 83-2735 831100 AIAA, AHS, IES, SETP, SFTE, and DGLR, Flight Testing Conference, 2nd, Las Vegas, NV, Nov. 16-18, 1983. 9 p. AD(Deutsche Forschungs- und Versuchsanstalt fuer Luft- und Raumfahrt, Institut fuer Flugmechanik, Brunswick, West Germany) 9 p. Refs. 8

The intention of the paper is to present a flight test technique for the evaluation of helicopter flight dynamics. Emphasis is placed on the discussion of the whole flight test loop including flight task definition, test procedure, the role of the test pilot, data acquisition, and data analysis. The technique is illustrated by tests for the assessment of mission demands. The test mission was combined from maneuvers which are representative of NOE-flight profiles. In addition results which specify the required agility of helicopters in NOE-maneuvering are outlined. Author

Controlled Terms: *AIRCRAFT MANEUVERS / *CONTROLLABILITY / DATA ACQUISITION / DATA REDUCTION / *FLIGHT TESTS / HELICOPTER CONTROL / *HELICOPTER PERFORMANCE / MISSION PLANNING / *NAP-OF-THE-EARTH NAVIGATION / WORKLOADS (PSYCHOPHYSIOLOGY)

Reference no 27

84A11062 NASA ISSUE 1 Category 8

Mission-specific effects on helicopter flight mechanics

Missionsspezifische Einfluesse auf die^Hubschrauber-Flugmechanik

HAMEL, P. G.; GMELIN, B. L.; PAUSDER, H.-J.

830000 IN: International Helicopter Forum, 14th, Bueckeburg and Hanover, West Germany, May 20, 21, 1982, Proceedings. Part 1 (A84-11051 01-01). Hanover, West Germany, Deutsche Messe- und Ausstellungs-AG, 1983, 27 p. In German. AC(Deutsche Forschungs- und Versuchsanstalt fuer Luft- und Raumfahrt, Institut fuer Flugmechanik, Brunswick, West Germany) 27 p. Refs. 18 Jpn. 18

It has been found that the evaluation of the flight characteristics of a helicopter depends to a large degree on the mission which has to be conducted. Questions can, for instance, arise regarding the design of criteria concerning the helicopter flight characteristics for a successful mission involving the defense against tanks. The present investigation has the objective to report the results of flight tests which had been conducted to determine the mission-oriented flight properties of helicopters. The obtained data provide a contribution to a data bank which has recently been established in West Germany as a basis for the derivation of guidelines regarding the flight characteristics of future helicopters. G.R.

Controlled Terms: AERODYNAMIC STABILITY / AIRCRAFT MANEUVERS / COMBAT / *FLIGHT CHARACTERISTICS / *FLIGHT MECHANICS / *FLIGHT TESTS / *HELICOPTER DESIGN / *MILITARY HELICOPTERS / MISSION PLANNING

Reference no 28

84A10144 NASA ISSUE 1 Category 2

Vortex and momentum theories for hovering rotors

FLAX, A. H.

831100 AIAA Journal (ISSN 0001-1452), vol. 21, Nov. 1983, p. 1595, 1596. AA(Institute for Defense Analyses, Alexandria, VA) 2 p. Refs. 5 Jpn. 6

A seeming paradox in the vortex theory for hovering rotors has been discussed by Miller (1982). However, this apparent paradox can be resolved by recognizing that the assumption of cylindrical shape of vortex sheets is only valid in the ultimate wake. It is shown that vortex theory can be unambiguously applied to the hovering rotor and, for the case of an infinite number of blades, is substantially equivalent to the momentum theory of the actuator disk. However, simplified models of the vortex wake such as those discussed by Miller are useful, since they can often provide valuable insights into aspects of vortex interaction with the rotor. G.R.

Controlled Terms: ACTUATOR DISKS / *FLOW THEORY / *HOVERING / *MOMENTUM THEORY / *ROTOR AERODYNAMICS / *VORTICES / WAKES

Reference no 29

83A50141 NASA ISSUE 24 Category 5

Aeromechanical stability of a hingeless rotor in hover and forward flight - Analysis and wind tunnel tests

YEAGER, W. T., JR.; MANTAY, W. R.; HAMOUDA, M.-N. H.

830900 Associazione Industrie Aerospaziali and Associazione Italiana di Aeronautica ed Astronautica, European Rotorcraft Forum, 9th, Stresa, Italy, Sept. 13-15, 1983, Paper. 20 p. AB(U.S. Army, Army Structures Laboratory, Hampton, VA) AC(Vigyan Research Associates, Inc., Hampton, VA) 20 p. Refs. 16 Jpn. 3548

The ground resonance of soft inplane hingeless rotors was modeled analytically and the results were compared with experimental data. The numerical model applied was the Comprehensive Analytical Model of Rotorcraft Aerodynamics and Dynamics (CAMRAD), which used as input the elastic degrees of freedom in flap bending, lead-lag bending and torsion, and a rigid pitch degree of freedom. The output described the elastic motion of the fuselage and rotor support system in the wind tunnel. Both flutter and trim analyses were performed. A soft inplane hingeless rotor with NACA 0012 blades was subjected to trials in the NASA Langley Transonic Dynamics Tunnel with Freon-12 as the test medium. The tests covered conditions of hover and forward flight. Data were gathered on the rotor lead-lag regressing mode damping. The model correctly predicted the ground resonance instability experienced in the hover trials, and the frequency and damping values of the lead-lag regressing mode in hover and forward flight. M.S.K.

Controlled Terms: *AERODYNAMIC STABILITY / COMPUTATIONAL FLUID DYNAMICS / DAMPING / *FLIGHT STABILITY TESTS / FLUTTER ANALYSIS / *HOVERING STABILITY / *RIGID ROTORS / *ROTOR AERODYNAMICS / TRANSONIC WIND TUNNELS / *WIND TUNNEL TESTS

Reference no 30

83A48365 NASA ISSUE 23 Category 5

NOTAR - The viable alternative to a tail rotor

SAMPATACOS, E. P.; MORGER, K. M.; LOGAN, A. H.

AIAA PAPER 83-2527 831000 American Institute of Aeronautics and Astronautics, Aircraft Design, Systems and Technology Meeting, Fort Worth, TX, Oct. 17-19, 1983. 9 p. AC(Hughes Helicopters, Inc., Culver City, CA) 9 p. Refs. 8 Jpn. 3404

A single rotor helicopter directional control system utilizing an enclosed fan has been developed by an American aerospace company. The NO Tail Rotor (NOTAR) system consists of a variable pitch fan, a circulation control tail boom, a valved turning vane array ('direct jet thruster'), and a vertical fin. It is pointed out that the NOTAR system separates the yaw moment required to trim the main rotor torque from the yaw moments the pilot uses to maneuver the aircraft. For a NOTAR technology demonstration, the tail rotor in a helicopter was replaced with the NOTAR system for ground and flight tests. The demonstration proved that the NOTAR system is a viable alternative to a tail rotor for single main rotor helicopters. It is found that the NOTAR system is inherently safer than a tail rotor helicopter. G.R.

Controlled Terms: *AIRCRAFT SAFETY / COMPUTERIZED SIMULATION / *CONTROLLABILITY / DIRECTIONAL CONTROL / *FLIGHT TESTS / *HELICOPTER CONTROL / *HELICOPTER DESIGN / HOVERING STABILITY / OH-6 HELICOPTER / *TAIL ROTORS / TORQUE / YAWING MOMENTS

Reference no 31

83A41920 NASA ISSUE 19 Category 8

On approximating higher-order rotor dynamics in helicopter stability-derivative models

HANSEN, R. S.

AIAA PAPER 83-2088 830800 American Institute of Aeronautics and Astronautics, Atmospheric Flight Mechanics Conference, Gatlinburg, TN, Aug. 15-17, 1983. 12 p. AA(U.S. Army, Aeromechanics Laboratory, Moffett Field, CA) 12 p. Refs. 7

A primary objective of helicopter flight-test stability- and control-derivative extraction is to obtain results that are consistent with the theoretical quasi-static derivatives. A previous investigation, in which a model was identified from simulation data using the conventional six-degree-of-freedom stability-derivative model structure, showed that identified derivatives can differ considerably from those obtained by theoretical analysis. The fundamental reason for these discrepancies has been attributed to the lack of adequate modeling of the higher-order rotor (particularly flapping) dynamics. In this paper, several possible methods for including rotor-dynamic effects without making unacceptable increases in either the order of the system or the number of parameters to be identified are reported. It is shown that approximating rotor-dynamic effects yields identified derivative values that are more consistent with the theoretical quasi-static derivatives. Although the application is directed toward the helicopter, the methodology utilized can also be applied to flexible aircraft, unsteady aerodynamics, and highly augmented flight vehicles. Author

Controlled Terms: AIRCRAFT STABILITY / *FLAPPING / FLEXIBLE WINGS / FLIGHT SIMULATION / *HELICOPTER CONTROL / HELICOPTER PERFORMANCE / *PARAMETER IDENTIFICATION / *ROTARY WINGS / *ROTOR AERODYNAMICS / *STABILITY DERIVATIVES

Reference no 32

83A41907 NASA ISSUE 19 Category 8

Effects of rotor inertia and rpm control on helicopter handling qualities

CORLISS, L. D.; BLANKEN, C. L.; NELSON, K.

National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)

AIAA PAPER 83-2070 830800 American Institute of Aeronautics and Astronautics. Atmospheric Flight Mechanics Conference, Gatlinburg, TN, Aug. 15-17, 1983. 8 p. AA(NASA, Ames Research Center; U.S. Army, Aeromechanics Laboratory, Moffett Field, CA) AB(U.S. Army, Aeromechanics Laboratory, Moffett Field, CA) AC(Computer Sciences Corp., Moffett Field, CA) 8 p.

The effects of thrust-response characteristics on helicopter handling qualities have until recently remained largely undefined. A multiphase program is being conducted to study, in a generic sense and through ground simulation, the effects of engine and rotor response characteristics, excess power, and vertical damping on specific maneuvers included in nap-of-the-earth (NOE) operations. This paper describes the most recent of these phases: a simulation in which the effects on handling qualities of rotor inertia and rpm changes were considered. Thrust- and height-response characteristics to step-control inputs are included to document the configurations investigated. Results indicate that with a given engine response and unlimited power, large changes in rotor inertia have little effect on handling qualities for certain low-speed tasks and hover tasks. The effects on handling qualities of requiring the pilot to maintain proper rotor rpm limits were also studied. This investigation revealed that large fluctuations in rotor rpm degrade handling qualities; as a result, continued study of the use of methods to automate control of rotor rpm is recommended. Author

Controlled Terms: DYNAMIC RESPONSE / *HELICOPTER CONTROL / INERTIA / MILITARY HELICOPTERS / *ROTARY WINGS / *ROTOR AERODYNAMICS / SPEED CONTROL / SPEED REGULATORS / THRUST / VERTICAL MOTION SIMULATORS

Reference no 33

83A41078 NASA ISSUE 19 Category 8

The effects of engine and height-control characteristics on helicopter handling qualities

CORLISS, L. D.

830700 (American Helicopter Society and NASA Ames Research Center, Specialists' Meeting on Helicopter Handling Qualities, Moffett Field, CA, Apr. 14-15, 1982) American Helicopter Society, Journal (ISSN 0002-8711), vol. 28, July 1983, p. 56-62. AA(U.S. Army, Army Research and Technology Laboratories, Moffett Field, CA) 7 p. Refs. 11

A ground-based simulation study was conducted on a large-scale motion simulator to study the effects in the vertical axis of engine response characteristics on handling qualities for a nap-of-the-earth (NOE) operating environment. This study concentrated specifically on the helicopter configuration with a speed-governed gas-turbine and expands previous work by focusing on aspects peculiar to rotary-wing and NOE operations. A wide range of engine response time, vehicle damping and sensitivity and excess power levels was studied. The data are compared with the existing handling-qualities specifications, MIL-F-83300 and AGARD 577, and in general show a need for higher minimums when performing such NOE maneuvers as a dolphin and bob-up task. Author

Controlled Terms: *AIRCRAFT MANEUVERS / *CONTROLLABILITY / ELECTRONIC CONTROL / *ENGINE CONTROL / *FLIGHT SIMULATION / FUEL CONTROL / GAS TURBINE ENGINES / HEIGHT / HELICOPTER ENGINES / *HELICOPTER PERFORMANCE / *NAP-OF-THE-EARTH NAVIGATION

Reference no 34

83A32785 NASA ISSUE 14 Category 5

Aeroelastic stability of an elastic circulation control rotor blade in hover

CHOPRA, I.

AIAA PAPER 83-0985 830500 N00167-82-M-3701 AIAA, ASME, ASCE, and AHS, Structures, Structural Dynamics and Materials Conference, Lake Tahoe, NV, May 2-4, 1983. 10 p. AA(Maryland, University, College Park, MD) 10 p. Refs. 9

A finite element method (FEM) is modified to study the aeroelastic stability of a controlled circulation rotor (CCR) blade. Controlled circulation involves venting a thin jet through a slot on the rounded trailing edge in order to energize the boundary layer, create a Coanda effect, and thus delay separation. The FEM formulation consists of division of the blade into discrete beam elements, each with two end nodes and three internal nodes, yielding 15 degrees of freedom. The aerodynamic forces are calculated according to the airfoil characteristics for section lift, drag, and moment at the midchord. An iterative procedure is employed to obtain a nonlinear trim solution, while a flutter solution is calculated by assuming the blade motions to be small perturbations about the steady solution. The mode is used for various proposed CCR blades, showing that, e.g., soft inplane CCR blades have excessive external damping to stabilize the lag mode. Necessary levels of internal structural damping are defined for CCR design configurations. M.S.K.

Controlled Terms: AERODYNAMIC LOADS / *AEROELASTICITY / BLOWING / *CIRCULATION CONTROL ROTORS / FINITE ELEMENT METHOD / *HOVERING STABILITY / NONLINEAR SYSTEMS / *ROTOR AERODYNAMICS / TRAILING EDGES

Reference no 35

83A29865 NASA ISSUE 12 Category 5

Measured inplane stability characteristics in hover for an advanced bearingless rotor

WELLER, W. H.; WARMBRODT, W.

Bell Helicopter Co., Fort Worth, Tex. (BI370951)

AIAA 83-0987 830000 IN: Structures, Structural Dynamics and Materials Conference, 24th, Lake Tahoe, NV, May 2-4, 1983, Collection of Technical Papers, Part 2 (A83-29729 12-39). New York, American Institute of Aeronautics and Astronautics, 1983, p. 536-546. AA(Bell Helicopter Textron, Fort Worth, TX) AB(NASA, Ames Research Center, Moffett Field, CA) 11 p. Refs. 11

A study was made of the inplane stability characteristics of a model of an advanced bearingless main rotor. The four-bladed rotor system was tested for aeroelastic stability in hover for both isolated and body-free conditions. Variations in several hub design parameters were tested, including blade coning and sweep angles, blade inplane structural damping, pitch-link location (delta-free effect), and fuselage structural damping. An analysis of the experimental results shows that parametric stability trends observed from isolated-rotor studies may yield incorrect conclusions regarding coupled rotor-body characteristics. For body-free conditions, the baseline rotor configuration resulted in the best stability margins at 1-g thrust as well as at the nominal design rotor speed. Blade built-in coning had little effect on the body-free damping levels at the inplane/body resonance points. Introducing blade sweep destabilized the rotor at the inplane/body pitch resonance. V.L.

Controlled Terms: *AERODYNAMIC STABILITY / AERDELASTICITY / *AIRCRAFT MODELS / *BEARINGLESS ROTORS / FUSELAGES / HELICOPTER DESIGN / *HOVERING STABILITY / HUBS / PITCH (INCLINATION) / ROLL / *ROTOR AERODYNAMICS / *STABILITY TESTS / SWEEP ANGLE / VIBRATION DAMPING

Reference no 36

83A29864 NASA ISSUE 12 Category 5

A closed-form analysis of rotor blade flap-lag stability in hover and low-speed forward flight in turbulent flow

PRUSSING, J. E.; LIN, Y. K.

830000 DAAG29-81-K-0072 IN: Structures, Structural Dynamics and Materials Conference, 24th, Lake Tahoe, NV, May 2-4, 1983, Collection of Technical Papers. Part 2 (A83-29729 12-39). New York, American Institute of Aeronautics and Astronautics, 1983, p. 529-535. AB(Illinois, University, Urbana, IL) 7 p. Refs. 10

An approximate, closed-form analysis is conducted for the stability of the coupled flap-lag motion of a helicopter rotor blade in the presence of turbulence. The longitudinal, lateral, and vertical components of turbulence are modelled as uncorrelated physical white noise processes and the equations governing the first moments of the stochastic averaging method. Approximations based on realistic rms values of turbulence velocities are used to obtain a closed-form first moment stochastic stability criterion. Results show that in hover the turbulence increases the stability of the coupled flap-lag motion. It is suggested that the turbulence increase the damping in the least-stable, lead-lag mode by providing the same stabilizing effect as an increase in the profile drag coefficient in both hover and low-speed forward flight. The vertical turbulence is found to have the dominant effect on this increase in stability. Numerical results for the hover case are compared with the more stringent case of second moment stability. N.B.

Controlled Terms: *AERODYNAMIC STABILITY / AERDELASTICITY / AIRCRAFT STABILITY / ELASTIC DAMPING / *FLAPPING / HORIZONTAL FLIGHT / *HOVERING / RIGID ROTORS / *ROTARY WINGS / *ROTOR AERODYNAMICS / STABILITY DERIVATIVES / STOCHASTIC PROCESSES / TURBULENCE EFFECTS / *TURBULENT FLOW

Reference no 37

83N28000# NASA ISSUE 17 Category 8

Calculation of the longitudinal stability derivatives and modes of motion for helicopter aircraft

M.S. Thesis

ONEILL, H. J.

Naval Postgraduate School, Monterey, Calif. (NS368219)

AD-A125593 821000 114 p. Jpn. 2695 HC A06/MF A01

This thesis presents an analysis of the longitudinal stability derivatives for helicopter aircraft and is intended to be used as a resource document for a helicopter stability and control course at the Naval Postgraduate School. Emphasis is given to the evolution of forces and moments on the helicopter, calculation of the stability derivatives at high advance ratios, derivation of the stability determinant and solution of the characteristic equation to yield the modes of motion of the helicopter. Author (GRA)

Controlled Terms: *AXES OF ROTATION / CALCULATORS / COMPUTER PROGRAMS / EQUATIONS OF MOTION / *HELICOPTERS / LOADS (FORCES) / *LONGITUDE / MANEUVERS / MANUAL / MOMENTS / *MOTION STABILITY

Reference no 38

83N23275# NASA ISSUE 13 Category 2

Effects of aerodynamic interaction between main and tail rotors on helicopter hover performance and noise

MENGER, R. P.; WOOD, T. L.; BRIEGER, J. T.

Textron Bell Helicopter, Fort Worth, Tex. (TV739419)

NASA-CR-166477; NAS 1.26:166477 830200 NAS2-10771 343 p. Jpn. 1988 HC A15/MF A01

A model test was conducted to determine the effects of aerodynamic interaction between main rotor, tail rotor, and vertical fin on helicopter performance and noise in hover out of ground effect. The experimental data were obtained from hover tests performed with a .151 scale Model 222 main rotor, tail rotor and vertical fin. Of primary interest was the effect of location of the tail rotor with respect to the main rotor. Penalties on main rotor power due to interaction with the tail rotor ranged up to 3% depending upon tail rotor location and orientation. Penalties on tail rotor power due to fin blockage alone ranged up to 10% for pusher tail rotors and up to 50% for tractor tail rotors. The main rotor wake had only a second order effect on these tail rotor/fin interactions. Design charts are presented showing the penalties on main rotor power as a function of the relative location of the tail rotor. S.L.

Controlled Terms: AERODYNAMIC CONFIGURATIONS / *AERODYNAMIC INTERFERENCE / *AIRCRAFT NOISE / *FINS / *HELICOPTER TAIL ROTORS / *HOVERING / *ROTOR AERODYNAMICS / SCALE MODELS / TABLES (DATA) / TEST FACILITIES

Reference no 39

83N22205# NASA ISSUE 12 Category 8

Efficient algorithms for computing trim and small-disturbance equations of motion of aircraft coordinated and uncoordinated, steady, steep turns

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National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)

NASA-TM-84324; A-9220; NAS 1.15:84324 830200 41 p. Jpn. 1831 HC A03/MF A01

The development of computational algorithms that permit efficient calculation of aircraft trim states and of the associated small disturbance equations of motion for a systematic investigation of the statics and dynamics of aircraft in coordinated and uncoordinated, steady, steep turning flight is reported. The efficiency in the trim computation is realized by decoupling the governing equations. The small disturbance equations of motion, which are given in a general body axis system, include aerodynamic acceleration derivatives; they are cast in a familiar first order, vector matrix format of modern system theory. These algorithms were applied to a variety of rotorcraft simulation models. Results pertaining to a simulated hingeless rotor helicopter are also presented S.L.

Controlled Terms: *AERODYNAMIC BALANCE / ALGORITHMS / COMPUTERIZED SIMULATION / *DYNAMIC CHARACTERISTICS / EQUATIONS OF MOTION / *HELICOPTER PERFORMANCE / KINEMATICS / *RIGID ROTORS / *TURNING FLIGHT

Reference no 40

83N19711# NASA ISSUE 10 Category 2

Free wake techniques for rotor aerodynamic analysis. Volume 1: Summary of results and background theory

MILLER, R. H.

Massachusetts Inst. of Tech., Cambridge. (MJ700802)

NASA-CR-166434; NAS 1.26:166434; ASRL-TR-199-1 821200 NAG2-47 48 p. Jpn. 1458 HC A03/MF A01

Results obtained during the development of a consistent aerodynamic theory for rotors in hovering flight are discussed. Methods of aerodynamic analysis were developed which are adequate for general design purposes until such time as more elaborate solutions become available, in particular solutions which include real fluids effects. Several problems were encountered in the course of this development, and many remain to be solved, however it is felt that a better understanding of the aerodynamic phenomena involved was obtained. Remaining uncertainties are discussed. Author

Controlled Terms: *AERODYNAMIC CHARACTERISTICS / FLUID FLOW / *HOVERING / PROBLEM SOLVING / RESEARCH MANAGEMENT / *ROTOR AERODYNAMICS / *WAKES

Reference no 41

83A18380 NASA ISSUE 6 Category 5

Rotor blade flap-lag stability in turbulent flows

PRUSSING, J. E.; LIN, Y. K.

820400 DAAG29-78-0039 American Helicopter Society, Journal, vol. 27, Apr. 1982, p. 51-57. AB(Illinois, University, Urbana, IL) 7 p. Refs. 15

The stability of coupled flap-lag motion of a helicopter rotor blade in the presence of turbulence is investigated. The rigid blade flap-lag equations of motion previously derived by Peters are generalized to include random turbulence in the airspeed components of the blade. By assuming white noise turbulence and applying a special case of the stochastic averaging procedure, the equations are converted to Ito type stochastic differential equations which are then used to determine the stochastic stability of the system. Illustrated numerical results for the second moment (mean-square) stability boundaries are presented for the restricted case of a rotor blade in hover with vertical turbulence. The restriction to vertical turbulence is made because the vertical component is found to have a much larger effect on the coupled flap-lag motion than the in-plane components. The surprising result is obtained that the vertical-turbulence appears to increase the stability of the coupled flap-lag motion for realistic rms turbulence velocities. (Author)

Controlled Terms: AIRSPEED / *COMPUTATIONAL FLUID DYNAMICS / DIFFERENTIAL EQUATIONS / *EQUATIONS OF MOTION / *HOVERING STABILITY / RIGID ROTORS / *ROTARY WINGS / *STOCHASTIC PROCESSES / *TURBULENT FLOW / WHITE NOISE

Reference no 42

83N17531# NASA ISSUE 8 Category 5

A control model for maneuvering flight for application to a computer flight testing program

HAVERDINGS, H.

National Aerospace Lab., Amsterdam (Netherlands). (NE736790)

NLR-MP-81046-U 810731 Flight Div. Sponsored by Royal Netherlands Air Force Directorate of Material Presented at 7th European Rotorcraft and Powered Lift Aircraft Forum, Garmisch-Partenkirchen, West Ger., 8-11 Sep. 1987 18 p. Jpn. 1134 HC A02/MF A01

A computer-flight-testing (CFT) proaram for helicopters was developed to evaluate helicopter dynamics and handling and control qualities. The nonlinear 6 degrees of freedom helicopter model is driven by control inputs by a specially developed control model (or pseudo pilot). This is an adaptation of a linear optimal control model as used in human factor analysis. The helicopter model is based on two dimensional strip aerodynamics and steady-state rotor blade dynamics using only out-of-plane bending mode shapes, which are suitable for various types of rotor articulation. The pilot model consists of a flight path generation (FPG) model and a stabilization (STAB) model. The FPG model is based on linearized system dynamics using terminal optimal control, generating both the required flight path and the control inputs to achieve it. These controls are input into the helicopter model. The two flight paths are compared, and differences are fed back to the STAB model to generate corrective control inputs of such a nature that the helicopter-model-generated flight path tracks the required flight path generated by the FPG model. Also the STAB model is based on linearized system dynamics. As an example, two flare maneuvers are 'flown', and the results discussed. The pseudo-pilot model merforms well, provided that helicopter dynamics do not change much during a specific maneuver. J.M.S.

Controlled Terms: COMPUTERIZED SIMULATION / *CONTROLLABILITY / *DYNAMIC MODELS / *FLIGHT SIMULATION / FLIGHT TESTS / *HELICOPTER CONTROL / MAN MACHINE SYSTEMS / *MANEUVERABILITY / OPTIMAL CONTROL

Reference no 43

83N15267# NASA ISSUE 6 Category 2

Response studies of rotors and rotor blades with application to aeroelastic tailoring

Final Report, 1978 - 1982

FRIEDMANN, P. P.

California Univ., Los Angeles. (CD146017)

NASA-CR-169740; NAS 1.26:169740 821200 NSG-1578 Dept. of Mechanics and Structures. 17 p. Jpn. 792 HC A02/MF A01

Various tools for the aeroelastic stability and response analysis of rotor blades in hover and forward flight were developed and incorporated in a comprehensive package capable of performing aeroelastic tailoring of rotor blades in forward flight. The results indicate that substantial vibration reductions, of order 15-40%, in the vibratory hub shears can be achieved by relatively small modifications of the initial design. Furthermore the optimized blade can be up to 20% lighter than the original design. Accomplishments are reported for the following tasks: (1) finite element modeling of rotary-wing aeroelastic problems in hover and forward flight; (2) development of numerical methods for calculating the aeroelastic response and stability of rotor blades in forward fight; (3) formulation of the helicopter air resonance problem in hover with active controls; and (4) optimum design of rotor blades for vibration reduction in forward flight. A.R.H.

Controlled Terms: ACTIVE CONTROL / *AEROELASTICITY / DIFFERENTIAL EQUATIONS / *DYNAMIC STABILITY / *EQUILIBRIUM EQUATIONS / FINITE ELEMENT METHOD / GALERKIN METHOD / HOVERING / RIGID ROTORS / *ROTARY WINGS / *ROTOR AERODYNAMICS / TIME DEPENDENCE / *VIBRATION DAMPING

Reference no 44

83N14137# NASA ISSUE 5 Category 8

Dynamic stability of flight vehicles

M.S. Thesis

POULIEZOS, D. P.

Naval Postgraduate School, Monterey, Calif. (NS368219)

AD-A119637 820600 257 p. Jpn. 623 HC A12/MF A01

The thesis presents an analytical treatment of the dynamics of the flight vehicle and might be used as a textbook for a Dynamic Stability and Control advanced class. Concentration is given to derivation of equations of motion, investigation of particular modes of motion, stability derivatives, aerodynamic transfer functions and digital computer solutions. Author (GRA)

Controlled Terms: *AERODYNAMIC STABILITY / *AIRCRAFT CONFIGURATIONS / DEGREES OF FREEDOM / *DYNAMIC CONTROL / DYNAMIC RESPONSE / EQUATIONS OF MOTION / MISSILE TRAJECTORIES / *ROTARY WING AIRCRAFT / *STABILITY DERIVATIVES / TRANSFER FUNCTIONS

Reference no 45

83N13114# NASA ISSUE 4 Category 8

The effects of pilot stress factors on handling quality assessments during US/German helicopter agility flight tests

PAUSDER, H. J.; GERDES, R. M.

National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)

NASA-TM-84194; A-9084; NAS 1.15:84194 821000 AA(Deutsche Forschungs-und Versuchsanstalt fuer Luft-und Raumfahrt e.V., Braunschweig, West Germany) 20 p. Jpn. 475 HC A02/MF A01

Flight tests were conducted with two helicopters to study and evaluate the effects of helicopter characteristics and pilot and task demands on performance in nap-of-the-Earth flight. Different, low-level slalom courses were set up and were flown by three pilots with different levels of flight experience. A pilot rating questionnaire was used to obtain redundant information and to gain more insight into factors that influence pilot ratings. The flight test setups and procedures are described, and the pilot ratings are summarized and interpreted in close connection with the analyzed test data. Pilot stress is discussed. The influence of demands on the pilot, of the helicopter characteristics, and of other stress factors are outlined with particular emphasis on how these factors affect handling-qualities assessment. Author

Controlled Terms: *CONTROLLABILITY / DATA ACQUISITION / *FLIGHT STRESS (BIOLOGY) / *FLIGHT TESTS / *HELICOPTER PERFORMANCE / INTERNATIONAL COOPERATION / NASA PROGRAMS / *PILOT PERFORMANCE / STRESS (PSYCHOLOGY) / TASK COMPLEXITY

Reference no 46

83A12097 NASA ISSUE 2 Category 5

Military potential of the ABC

PRICE, G.

821200 Vertiflite, vol. 28, Nov.-Dec. 1982, p. 12-14. AA(United Technologies Corp., Sikorsky Aircraft Div., Stratford, CT) 3 p.

Helicopters incorporating the Advancing Blade Concept (ABC) coaxial hingeless rotor design are capable of increases in speed and altitude of 100 knots and 10,000 feet, respectively, over comparable conventional designs. Due to the use of stiff, hingeless rotors, ABC helicopters retain high agility and maneuverability throughout their speed range, and present the possibility of superior gun platform stability. The XH-59A technology demonstration aircraft has completed its test program, achieving a maximum speed of 260 knots and an altitude of over 25,000 feet. Attention is given to the future military requirements addressed by the modification of the present demonstration aircraft to a ducted pusher propeller configuration, with a propulsion system that incorporates a new main gearbox and two T700 turboshaft engines. O.C.

Controlled Terms: *AIRCRAFT CONFIGURATIONS / *AIRCRAFT MANEUVERS / FLIGHT TESTS / *HELICOPTER DESIGN / HELICOPTER ENGINES / *HELICOPTER PERFORMANCE / MANEUVERABILITY / *MILITARY AIRCRAFT / *RIGID ROTORS

Reference no 47

83N10068# NASA ISSUE 1 Category 8

Flight experiments with integrated isometric side-arm controllers in a variable stability helicopter

SINCLAIR, M.

National Aeronautical Establishment, Ottawa (Ontario). (NC052609)

820600 Flight Research Lab. In AGARD Criteria for Handling Qualities of Mil. Aircraft 9 p (SEE N83-10054 01-08) 9 p. Jpn. 12 HC A14/MF A01

The suitability of integrated, multiaxis, isometric controllers for use in helicopters were investigated. The 3 axis and 4 axis isometric side arm control configurations were flown successfully through a wide variety of demanding visual flight tasks and a brief instrument flight precision approach evaluation. The experimental tasks, the evaluated controller arrangements and the developed control laws are described, and the results of comparative assessments between isometric side arm control and conventional control arrangements are presented. E.A.K.

Controlled Terms: *AXES OF ROTATION / *CONTROLLABILITY / *FLIGHT CONTROL / FLIGHT SIMULATORS / HANDLES / *HELICOPTERS / *MAN MACHINE SYSTEMS / MANUAL CONTROL / *PILOTS (PERSONNEL) / *SEATS / SITTING POSITION

Reference no 48

83N10067# NASA ISSUE 1 Category 8

Operational criteria for the handling qualities of combat helicopters

STEWARD, W.

Army Air Corps, Stockbridge (England). (AY878010)

820600 In AGARD Criteria for Handling Qualities of Mil. Aircraft 5 p (SEE N83-10054 01-08) 5 p. Refs. 1 Jpn. 12 HC A14/MF A01

To minimize the threat from air and ground based weapon systems, combat helicopter operations require the use of concealed low level flight. The tasks facing the combat helicopter pilot during a typical antiarmor mission are discussed. Primary consideration is given to daylight operations in VMC, but the requirements for missions at night and in adverse weather, and for training are also addressed, together with the implications for handling qualities posed by the threat of armed helicopters in the air to air role. It is concluded that, by reducing the flying workload, assisting in the exploitation of maximum aircraft performance, and enhancing control accuracy, better handling qualities can contribute to improved operational effectiveness. E.A.K.

Controlled Terms: ANTIAIRCRAFT MISSILES / COMBAT / *CONTROLLABILITY / *HUMAN FACTORS ENGINEERING / *MANEUVERABILITY / *MILITARY HELICOPTERS / MILITARY OPERATIONS / NAP-OF-THE-EARTH NAVIGATION / PILOTS (PERSONNEL) / *TERRAIN FOLLOWING AIRCRAFT / *WEAPON SYSTEMS

Reference no 49

83N10066# NASA ISSUE 1 Category 8

The impact of active control on helicopter handling qualities

KRETZ, M.

Giravions Dorand Co., Suresnes (France). (GZ938173)

820600 In AGARD Criteria for Handling Qualities of Mil. Aircraft 10 p (SEE N83-10054 01-08) In FRENCH and in ENGLISH 10 p. Jpn. 11 HC A14/MF A01

Changes have occurred in the concepts of controlling the working conditions of helicopter rotors. New trends, prompted by the techniques of active control applied to fixed wings, are oriented towards the automatic control of dynamic phenomena (vibration, instability) and aerodynamic phenomena (stall effects, interaction, gusting). The new trends feature frequency responses much wider than those of conventional autopilots, extending up to 30 Hz. Military helicopter design is much more demanding with regard to handling qualities: higher disk loading, NOE mission requirement, advent of advanced rotor aircraft concepts and a general broadening of the flight envelope. The impact of active control on handling qualities and its benefits in their implementation are analyzed. It is concluded that present day control system limitations due to the use of monocyclic swashplate principles will have to be removed in the future by unconventional control systems based on multiloop self adaptive control resulting in higher order optimization of handling qualities. E.A.K.

Controlled Terms: *ACTIVE CONTROL / *AERODYNAMIC CHARACTERISTICS / AUTOMATIC CONTROL / *CONTROLLABILITY / DYNAMIC STRUCTURAL ANALYSIS / FIXED WINGS / *MANEUVERABILITY / *MILITARY HELICOPTERS / ROTARY WINGS / ROTOR AERODYNAMICS

Reference no 50

83N10065# NASA ISSUE 1 Category 8

The status of military helicopter handling qualities criteria

KEY, D. L.

Army Research and Technology Labs., Moffett Field, Calif. (A2025071)

820600 Aeromechanics Lab. In AGARD Criteria for Handling Qualities of Mil. Aircraft 9 p (SEE N83-10054 01-08) 9 p. Jpn. 11 HC A14/MF A01

Current helicopter specifications were assessed. It is indicated that MIL-F-83300 has clear advantages in its broad coverage of important handling qualities aspects and its systematic structure. Its disadvantages are that it is primarily based on V/STOL data, and explicit helicopter characteristics are only lightly covered. The deficiencies resulted in a major effort to develop a new specification containing mission oriented handling qualities requirements. A revised specification for adoption as MIL-H-8501B is planned. E.A.K.

Controlled Terms: AIRCRAFT / AIRCRAFT LANDING / *CONTROLLABILITY / HELICOPTER PERFORMANCE / *LONGITUDINAL CONTROL / *MANEUVERABILITY / *MILITARY HELICOPTERS / MILITARY TECHNOLOGY / *PITCH (INCLINATION) / TAKEOFF / V/STOL AIRCRAFT

Reference no 51

83N10056# NASA ISSUE 1 Category 8

Status of VTOL and VSTOL flying qualities criteria development: Where are we and where are we going?

CLARK, J. W., JR.; GOLDSTEIN, K. W.

Naval Air Development Center, Warminster, Pa. (N0000154)

820600 Aircraft and Crew Systems Technology Directorate. In AGARD Criteria for Handling Qualities of Mil. Aircraft 18 p (SEE N83-10054 01-08) 18 p. Jpn. 10 HC A14/MF A01

Over the past decade, a number of weaknesses and omissions were uncovered in the VSTOL and Helicopter Flying Qualities Specifications (MIL-F-83300 and MIL-H-8501A). Identification of these weaknesses spawned technology development in a number of areas. Both interim and final results in some of these areas, the status of existing data bases, and the future criteria development needs as perceived by the US Navy are presented. Specific areas addressed include: (1) information display and IMC (Instrument Meteorological Conditions) flight requirements; (2) criteria definition for highly augmented, multi-mode control schemes; (3) requirements unique to the small seaborne platform operational environment; and (4) requirements unique to varied rotor configurations. Both fixed-wing and rotary-wing criteria are considered. A.R.H.

Controlled Terms: COMMAND AND CONTROL / *CONTROLLABILITY / *CRITERIA / DISPLAY DEVICES / DYNAMIC RESPONSE / EQUIVALENCE / *FLIGHT CHARACTERISTICS / *HELICOPTER CONTROL / HOVERING STABILITY / NAVY / *SPECIFICATIONS / SYSTEMS ANALYSIS / *V/STOL AIRCRAFT

Reference no 52

83N10054# NASA ISSUE 1 Category 8

Criteria for Handling Qualities of Military Aircraft

Advisory Group for Aerospace Research and Development, Neuilly-Sur-Seine (France). (AD455458)

AGARD-CP-333; ISBN-92-835-0313-8 820600 Symp. held in Fort Worth, Colo., 19-22 Apr. 1982 317 p. Jpn. 10 HC A14/MF A01

The status of flying qualities criteria for CTOL, V/STOL, and VTOL aircraft is reviewed and current and advanced flight control design techniques and handling quality requirements are examined with attention given to specifications for military aircraft. For individual titles, see N83-10055 through N83-10080

Controlled Terms: ACTIVE CONTROL / *AIRCRAFT CONTROL / AIRCRAFT PILOTS / ANGLE OF ATTACK / *CONTROLLABILITY / *CRITERIA / *FIGHTER AIRCRAFT / *FLIGHT CHARACTERISTICS / MATHEMATICAL MODELS / *MILITARY HELICOPTERS / V/STOL AIRCRAFT

Reference no 53

82A41687 NASA ISSUE 21 Category 8

A restrained model helicopter, which is able to fly, for investigations regarding human multiparameter control behavior --- German thesis

Ein flugfaehig eingespannter Modellhubschrauber^fuer Untersuchungen zum menschlichen^Mehrgroessenreglerverhalten

OESTERLIN, W. H.

800000 Darmstadt, Technische Hochschule, Fachbereich Regelungs- und Datentechnik, Dr.-Ing. Dissertation, 1980. 259 p. In German. Research supported by the Deutsche Forschungsgemeinschaften. 259 p. Refs. 77

The control of a Remotely Piloted Vehicle (RPV) with six degrees of freedom of motion represents for the human operator a very difficult problem. The present investigation is concerned with this problem. The conducted studies involve the control of the hovering flight of a remotely piloted model helicopter. A 'model helicopter trainer' was employed in place of a model engaged in free flight. By means of a special mechanism, the helicopter is connected to a carriage which is mounted on two rails. The constructional design involved permits helicopter hovering flight operations. The conduction of investigations regarding the control behavior of man requires that the dynamic behavior of the controlled system is known. A model for the operational characteristics of the helicopter in hovering flight was, therefore, developed. The model provided the basis for the development of a dynamic multiparameter control model for two experienced pilots. G.R.

Controlled Terms: *AIRCRAFT MODELS / BEARING (DIRECTION) / DYNAMIC MODELS / EXPERIMENTAL DESIGN / FLIGHT SIMULATION / GYROCOMPASSES / *HELICOPTER CONTROL / *HOVERING / *MAN MACHINE SYSTEMS / *PILOT PERFORMANCE / POSITION (LOCATION) / *REMOTELY PILOTED VEHICLES / ROTARY WINGS / SPEED CONTROL

Reference no 54

82A40946 NASA ISSUE 20 Category 2

An experimental and numerical study of 3-D rotor wakes in hovering flight

NSI MBA, M.; FAVIER, D.; MARESCA, C.

820000 DRET-78-456 In: International Council of the Aeronautical Sciences, Congress, 13th and AIAA Aircraft Systems and Technology Conference, Seattle, WA, August 22-27, 1982, Proceedings. Volume 1. (A82-40876 20-01) New York, American Institute of Aeronautics and Astronautics, 1982, p. 689-699. Direction des Recherches, Etudes et Techniques AC(Aix-Marseille I, Universite, Marseille, France) 11 p. Refs. 12

Results of comparison between experimental and numerical studies on the 3-D wake of a hovering rotor are presented. The wind-tunnel investigation is conducted by means of X-hot wires and laser Doppler anemometry procedures to measure the 3-D velocity field under the rotor and to determine the tip vortex paths for several rotor configurations. Additional flow visualizations and rotor airloads coefficients are also carried out. The prediction model is based on the classical vortex theory with an empirically prescribed geometry of the wake. From the blade circulation distribution the rotor wake is represented by vortex lines which are allowed to freely adapt until a converged wake geometry is obtained. Then a new estimate of the blade circulation repartition can be deduced. The procedure is repeated, iterating until the compatibility between the adapted wake geometry and the blade circulation repartition is obtained. The validity range of the calculation model is deduced from comparison with experimental data obtained on instantaneous velocities and tip vortex paths, for different rotor parameters including solidy, number of blades, pitch angle, blade twist, and tip shape. (Author)

Controlled Terms: AERODYNAMIC CONFIGURATIONS / *COMPUTATIONAL FLUID DYNAMICS / FLOW VELOCITY / FLOW VISUALIZATION / *HELICOPTER WAKES / *HOVERING / PERFORMANCE PREDICTION / *ROTOR AERODYNAMICS / THREE DIMENSIONAL FLOW / *VELOCITY DISTRIBUTION / *WIND TUNNEL TESTS

Reference no 55

82A40506 NASA ISSUE 20 Category 2

Theory and application of optimum airloads to rotors in hover and forward flight

MOFFITT, R. C.; BISSELL, J. R.

820000 In: American Helicopter Society, Annual Forum, 38th, Anaheim, CA, May 4-7, 1982, Proceedings. (A82-40505 20-01) Washington, DC, American Helicopter Society, 1982, p. 1-12. (For individual items see A82-40506 to A82-40556) AB(United Technologies Corp., Sikorsky Aircraft Div., Stratford, CT) 12 p.

A method is derived and applied that predicts optimum lift distribution for rotors in hover and forward flight. A key feature of the method is that it is formulated in terms of a matrix equation that gives a direct solution when wake geometry is fixed. An evaluation of the Theodorsen optimum static propeller theory, conducted with the analysis, indicates that the theory is not rigorous. It is shown that the torque differential term omitted in that analysis is both finite and significant. With this term included, the optimum static propeller wake displacement velocity is not constant. An optimized Black Hawk twist distribution in hover is shown to closely approximate the classic inverse radius pitch distribution predicted by strip momentum theory. The resulting downwash, however, is constant only over the inner 75% of the radius and substantial reductions occur in the tip region. Predicted improvements in Black Hawk forward flight performance with optimized azimuthally varying twist are significant but the associated variable twist is complex. (Author)

Controlled Terms: *AERODYNAMIC LOADS / DOWNWASH / FIXED WINGS / HELICOPTER WAKES / *HOVERING / LOAD DISTRIBUTION (FORCES) / OPTIMIZATION / *ROTARY WINGS / *ROTOR AERODYNAMICS / ROTOR LIFT

Reference no 56

82A39117 NASA ISSUE 19 Category 8

Flight dynamics of rotorcraft in steep high-g turns

CHEN, R. T. N.

National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)

AIAA PAPER 82-1345 820800 American Institute of Aeronautics and Astronautics, Atmospheric Flight Mechanics Conference, 9th, San Diego, CA, Aug. 9-11, 1982, 16 p. AA(NASA, Ames Research Center, Moffett Field, CA) 16 p. Refs. 21

An analyticl procedure developed to permit a systematic examination of rotorcraft flight dynamics in steep high-g turns is presented. The procedure is used in a numerical investigation of a tilt-rotor aircraft and three single-rotor helicopters that have different types of main rotor systems. The results indicate (1) that strong coupling in longitudinal and lateral-directional motions exists for these rotorcraft in high-g turns; (2) that for single-rotor helicopters, the direction of turn has a significant influence on flight dynamics; and (3) that a stability and control augmentation system that is designed on the basis of standard small-disturbance equations of motion from steady straight and level flight and that otherwise performs satisfactorily in operations near 1 g, becomes significantly degraded in steep turning flight. (Author)

Controlled Terms: AERODYNAMIC BALANCE / *AIRCRAFT CONTROL / AIRCRAFT DESIGN / AIRCRAFT STABILITY / EQUATIONS OF MOTION / *HELICOPTER PERFORMANCE / *HIGH GRAVITY ENVIRONMENTS / *MILITARY HELICOPTERS / NUMERICAL ANALYSIS / RIGID ROTORS / *ROTOR AERODYNAMICS / *TILT ROTOR AIRCRAFT / *TURNING FLIGHT

Reference no 57

82A38942 NASA ISSUE 19 Category 8

Simulator investigations of various side-stick controller/stability and control augmentation systems for helicopter terrain flight

AIKEN, E. W.

AIAA 82-1522 820000 In: Guidance and Control Conference, San Diego, CA, August 9-11, 1982, Collection of Technical Papers. (A82-38926 19-18) New York, American Institute of Aeronautics and Astronautics, 1982, p. 152-164. AA(U.S. Army, Aeromechanics Laboratory, Moffett Field, CA) 13 p. Refs. 15

Two piloted simulator experiments were conducted to assess the effects of side-stick-controller characteristics and level of stability and control augmentation on handling qualities for helicopter terrain flight. A composite of several evaluation tasks was flown with the aid of a head-up display of flight-control symbology. Variations in force-deflection characteristics and the number of axes controlled through a side-stick were investigated. Satisfactory handling qualities were achieved with a two-axis displacement controller and angular rate stabilization. Attitude stabilization was required to maintain adequate handling qualities for either a three- or four-axis rigid controller. (Author)

Controlled Terms: *AIRCRAFT STABILITY / *CONTROL STICKS / *CONTROLLABILITY / FLIGHT CHARACTERISTICS / FLIGHT CONTROL / *FLIGHT SIMULATORS / HEAD-UP DISPLAYS / *HELICOPTER CONTROL / NAP-OF-THE-EARTH NAVIGATION / *TERRAIN FOLLOWING AIRCRAFT

Reference no 58

82A37785 NASA ISSUE 18 Category 8

A simple system for helicopter individual-blade-control and its application to stall-induced vibration alleviation

HAM, N. D.; QUACKENBUSH, T. R.

Massachusetts Inst. of Tech., Cambridge. (MJ700802)

AHS PREPRINT 81-12 811100 American Helicopter Society, Northeast Region National Specialists' Meeting on Helicopter Vibration: Technology for the Jet Smooth Ride, Hartford, CT, Nov. 2-4, 1981, 10 p. NASA-sponsored research. AB(MIT, Cambridge, MA) 10 p.

A new, advanced type of active control for helicopters and its application to a system for stall flutter suppression is described. The system, based on previously developed M.I.T. Individual-Blade-Control hardware, employs blade-mounted accelerometers to sense torsional oscillations and feeds back rate information to increase the damping of the first torsion mode. A linear model of the blade and control system dynamics is used to give qualitative and quantitative guidance in the design process as well as to aid in analysis of experimental results. System performance in wind tunnel tests is described and evidence is given of the system's ability to provide substantial additional damping to stall-induced blade oscillations. (Author)

Controlled Terms: ACCELEROMETERS / *AERODYNAMIC STALLING / *AIRCRAFT STABILITY / *FEEDBACK CONTROL / FLUTTER / *HELICOPTER CONTROL / HELICOPTER DESIGN / ROTARY WINGS / *ROTOR BLADES / TORSIONAL VIBRATION / *VIBRATION DAMPING / WIND TUNNEL TESTS

Reference no 59

82A35818 NASA ISSUE 17 Category 2

Extension of local momentum theory to hovering rotor with distorted wake

KAWACHI, K.

820700 Journal of Aircraft, vol. 19, July 1982, p. 538-545. AA(Tokyo, University, Tokyo, Japan) 8 p. Refs. 19

The local momentum theory is extended in this paper to include the effects of a rotor wake deformation in hovering flight. A comparison of this extended local momentum theory with a prescribed wake vortex theory is presented. The results indicate that the extended local momentum theory has the capability of achieving a level of accuracy similar to that of the prescribed wake vortex theory, using a much smaller amount of computer time. It is also shown that the analytical results obtained using either theory are in reasonable agreement with experimental data. (Author)

Controlled Terms: COMPUTER PROGRAMS / *HELICOPTER WAKES / *HOVERING / LIFT / *MOMENTUM THEORY / *ROTOR AERODYNAMICS / RUN TIME (COMPUTERS) / *VORTICES

Reference no 60

82N33398# NASA ISSUE 24 Category 8

A ground-simulator investigation of helicopter longitudinal flying qualities for instrument approach

LEBACQZ, J. V.; FORREST, R. D.; GERDES, R. M.

National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)

NASA-TM-84225; A-8983; NAS 1.15:84225 820900 AB(FAA, Moffett Field, Calif.) 91 p. Jpn. 3393 HC A05/MF A01

A ground-simulation experiment was conducted to investigate the direct and interactive influences of several longitudinal static and dynamic stability parameters on helicopter flying qualities during terminal-area operations in instrument conditions. Variations that were examined included five levels of static control-position gradients ranging from stable to unstable; two levels of dynamic stability for the long-period oscillation; two levels of the steady-state pitch speed gradient; two levels of angle-of-attack stability and pitch-rate damping; and two levels of stability and control augmentation. These variations were examined initially in calm air and thin in simulated light-to-moderate turbulence and wind shear. Five pilots performed a total of 223 evaluations of these parameters for a representative microwave landing system precision approach task conducted in a dual-pilot crew-loading situation. Author

Controlled Terms: DYNAMIC STABILITY / *FLIGHT CHARACTERISTICS / *FLIGHT SIMULATORS / *GROUND BASED CONTROL / *HELICOPTER PERFORMANCE / *INSTRUMENT APPROACH / *LONGITUDINAL STABILITY / MICROWAVE EQUIPMENT / STATIC STABILITY

Reference no 61

82N32342# NASA ISSUE 23 Category 5

Aeroelastic stability of rotor blades using finite element analysis

CHOPRA, I.; SIVANERI, N.

Stanford Univ., Calif. (S0380476)

NASA-CR-166389; NAS 1.26:166389 820800 NCC2-13 Joint Inst. for Aeronautics and Acoustics. 119 p. Jpn. 3235 HC A06/MF A01

The flutter stability of flap bending, lead-lag bending, and torsion of helicopter rotor blades in hover is investigated using a finite element formulation based on Hamilton's principle. The blade is divided into a number of finite elements. Quasi-steady strip theory is used to evaluate the aerodynamic loads. The nonlinear equations of motion are solved for steady-state blade deflections through an iterative procedure. The equations of motion are linearized assuming blade motion to be a small perturbation about the steady deflected shape. The normal mode method based on the coupled rotating natural modes is used to reduce the number of equations in the flutter analysis. First the formulation is applied to single-load-path blades (articulated and hingeless blades). Numerical results show very good agreement with existing results obtained using the modal approach. The second part of the application concerns multiple-load-path blades, i.e. bearingless blades. Numerical results are presented for several analytical models of the bearingless blade. Results are also obtained using an equivalent beam approach wherein a bearingless blade is modelled as a single beam with equivalent properties. Results show the equivalent beam model. Author

Controlled Terms: AERODYNAMIC LOADS / *AEROELASTICITY / BEARINGLESS ROTORS / DEGREES OF FREEDOM / EQUATIONS OF MOTION / *FINITE ELEMENT METHOD / *FLUTTER ANALYSIS / *HELICOPTERS / *HOVERING / ITERATION / *ROTOR BLADES

Reference no 62

82N29312# NASA ISSUE 20 Category 5

Flap-lag-torsional dynamics of extensional and inextensional rotor blades in hover and in forward flight

Semiannual Progress Report, Jan. - Jun. 1982

DASILVA, C.

Cincinnati Univ., Ohio. (CP730085)

NASA-CR-169159; NAS 1.26:169159 820600 NAG2-38 Dept. of Aerospace Engineering and Applied Mechanics. 72 p. Jpn. 2788 HC A04/MF A01

The reduction of the O(cu epsilon) integro differential equations to ordinary differential equations using a set of orthogonal functions is described. Attention was focused on the hover flight condition. The set of Galerkin integrals that appear in the reduced equations was evaluated by making use of nonrotating beam modes. Although a large amount of computer time was needed to accomplish this task, the Galerkin integrals so evaluated were stored on tape on a permanent basis. Several of the coefficients were also obtained in closed form in order to check the accuracy of the numerical computations. The equilibrium solution to the set of 3n equations obtained was determined as the solution to a minimization problem. S.L.

Controlled Terms: COMPUTER PROGRAMS / DIFFERENTIAL EQUATIONS / *DYNAMIC CHARACTERISTICS / *FLAPS (CONTROL SURFACES) / GALERKIN METHOD / *HORIZONTAL FLIGHT / *HOVERING / INTEGRAL EQUATIONS / *ROTARY WINGS / *TORSION

Reference no 63
82A29042 NASA ISSUE 13 Category 5
Turbulence-excited flapping motion of a rotor blade in hovering flight
LIN, Y. K.; HONG, C. Y. R.
810000 DAAG29-78-G-0039 In: 1981 advances in aerospace structures and materials; Proceedings of the Winter Annual Meeting, Washington, DC, November 15-20, 1981. (A82-29026 13-31) New York, American Society of Mechanical Engineers, 1981, p. 149-153. AA(Illinois, University, Urbana, IL) 5 p. Refs. 10 Jpn. 2019

Statistical properties of a randomly vibrating rotor blade under the excitation of turbulence is investigated using a simple model which consists of a rigid blade, centrally hinged, with elastic restraint for the flapping motion. It is shown that turbulence affects the blade motion in two ways, distinguished by two types of terms in the equation of motion. The first type appears in the coefficients, thus causing the dynamic system to change randomly with time. The second type appears among the inhomogeneous terms. A theoretical procedure is proposed in which both effects of the turbulence can be accounted for. Application of the methodology is illustrated by numerical examples. (Author)

Controlled Terms: EQUATIONS OF MOTION / *FLAPPING HINGES / *HELICOPTERS / *HOVERING STABILITY / RANDOM VIBRATION / *ROTARY WINGS / STATISTICAL ANALYSIS / *TURBULENCE EFFECTS

Reference no 64
82N28285# NASA ISSUE 19 Category 5
Evaluations of helicopter instrument-flight handling qualities
SINCLAIR, S. R. M.; KERELIUK, S.
National Research Council of Canada, Ottawa (Ontario). (NL210403)
AD-A114004; NRCC-LR-608 820100 Flight Research Lab. In ENGLISH; FRENCH summary 49 p. Jpn. 2641 HC A03/MF A01

The NAE Airborne Simulator, a modified and suitably equipped Bell 205A helicopter, was used in experiments to provide background information on the handling qualities requirements for helicopter instrument flight. This investigation was in support of a regulatory review undertaken by the U.S. Federal Aviation Administration as part of an overall assessment of the helicopter certification process. The results illustrate the inter-dependence of the various stability and control characteristics which contribute to safe instrument flight handling qualities, and underline the importance of good mission simulation in conducting certification-related experiments. Author (GRA)

Controlled Terms: *AIRCRAFT CONTROL / AIRCRAFT SAFETY / AIRCRAFT STABILITY / *CONTROLLABILITY / DAMPING / FLIGHT SIMULATORS / *HELICOPTERS / *INSTRUMENT FLIGHT RULES

Reference no 65
82A26400 NASA ISSUE 11 Category 8
The Circulation Control Rotor /CCR/ control system
BARNES, D. R.
801000 American Helicopter Society, National Specialists Meeting on Rotor System Design, Philadelphia, PA, Oct. 22-24, 1980, Paper. 10 p. AA(Kaman Aerospace Corp., Bloomfield, CT) 10 p. Refs. 8 Jpn. 1708

This paper presents an overview of the XH-2/CCR, Circulation Control Rotor Flight Demonstrator, control system design. The circulation control rotor is a major departure from conventional rotor systems and presents the control system designer with both a new set of options and a new set of challenges. The paper emphasizes the CCR unique components and aspects of the control system. Component selection, sizing, integration and operation are addressed. Specific details of the XH-2/CCR control system are presented. Multicyclic control capability is included. (Author)

Controlled Terms: AERODYNAMIC CHARACTERISTICS / AIR FLOW / *CIRCULATION CONTROL ROTORS / COMPRESSORS / CONTROL EQUIPMENT / FLOW DISTRIBUTION / *HELICOPTER CONTROL / HELICOPTER DESIGN / MANUAL CONTROL / *ROTARY WINGS / *ROTOR AERODYNAMICS / SIZE DETERMINATION / STRUCTURAL MEMBERS / SYSTEMS ENGINEERING

Reference no 66
82A26396 NASA ISSUE 11 Category 5
Performance and aeroelastic tradeoffs on recent rotor blade designs
SCHRAGE, D. P.; OMALLEY, J. A.
801000 American Helicopter Society, National Specialists Meeting on Rotor System Design, Philadelphia, PA, Oct. 22-24, 1980, Paper. 11 p. AB(U.S. Army, Development and Qualification Directorate, St. Louis, MO) 11 p. Refs. 15 Jpn. 1704

A review is conducted of the performance and aeroelastic requirements of a helicopter rotor blade in both hover and forward flight. It is found that aeroelastic simplifications in rotor blade design are no longer practical if optimal performance is to be achieved. It is necessary to consider airfoil geometric variations. The influence of these variations on drag divergence Mach number at lift coefficients from near zero to near maximum lift and on pitching moments must be properly addressed during the design process. Blade torsional frequency, torsional stiffness, and section pitching moment, are exceedingly important parameters in optimizing performance and control loads. These parameters must, therefore, be adequately represented in analytical models. G.R.

Controlled Terms: AERODYNAMIC COEFFICIENTS / *AEROELASTICITY / *HELICOPTER DESIGN / *HOVERING STABILITY / LIFT / MACH NUMBER / PARAMETERIZATION / *PERFORMANCE PREDICTION / *PITCHING MOMENTS / *ROTARY WINGS / STIFFNESS / TORSIONAL STRESS

Reference no 67
82A26383 NASA ISSUE 11 Category 8
Selection of some rotor parameters to reduce pitch-roll coupling of helicopter flight dynamics
CHEN, R. T. N.
National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)
801000 American Helicopter Society, National Specialists Meeting on Rotor System Design, Philadephia, PA, Oct. 22-24, 1980, Paper. 19 p. AA(NASA, Ames Research Center, Moffett Field, CA) 19 p. Refs. 21 Jpn. 1708

The results of a study conducted to investigate further a means of choosing primary rotor parameters to reduce the coupling of longitudinal and lateral flapping in hover and in forward flight are presented. The rotor parameters included - flapping hinge offset, flapping hinge restraint, pitch-flap coupling, and blade Lock number - are known to influence the agility, stability, and operational safety of helicopters. Effects of the nonuniform downwash model of White and Blake on the blade flapping motion are examined, and the theoretical calculation is then correlated with experimental test data. The condition for achieving perfect decoupling of the flapping response due to aircraft pitch and roll rates, which was previously obtained for a hovering rotor, is evaluated in forward flight. The results show that negligible coupling is achieved in forward flight; moreover, there is the additional benefit of a slight reduction in the coupling of the roll rate to coning. It is also indicated that the values of the rotor parameters chosen according to the decoupling condition are moderate and that the flapping motion is stable with the parameters chosen. (Author)

Controlled Terms: AERODYNAMIC STABILITY / AIRCRAFT SAFETY / *DECOUPLING / *FLAPPING / FLAPPING HINGES / *HELICOPTER CONTROL / HOVERING / LATERAL CONTROL / LONGITUDINAL CONTROL / *PITCH (INCLINATION) / *ROLL / *ROTARY WINGS

Reference no 68

82A26227 NASA ISSUE 11 Category 2

The vortex flow field generated by a hovering helicopter

REDDY, K. R.

810000 In: Australasian Conference on Hydraulics and Fluid Mechanics, 7th, Brisbane, Australia, August 18-22, 1980, Preprints of Papers. (A82-26176 11-34) Barton, Australia, Institution of Engineers, 1981, p. 553-556. AA(Department of Defence, Aeronautical Research Laboratories, Melbourne, Australia) 4 p. Refs. 11 Jpn. 1699

Important vortex elements in the wake of a hovering helicopter are identified on the basis of well established fixed wing theory. The vortex positions are specified using experimental results. To make the problem mathematically simple, Willmer's rectangularisation principle is introduced. Using this wake model, the calculated velocity field and blade loading are compared with available helicopter flight data. The comparison shows that the present simple method yields satisfactory results. (Author)

Controlled Terms: COMPUTATIONAL FLUID DYNAMICS / COMPUTER TECHNIQUES / DROPS (LIQUIDS) / *FLOW DISTRIBUTION / FLOW THEORY / *HELICOPTER WAKES / *HOVERING / NUMERICAL ANALYSIS / *ROTOR AERODYNAMICS / SPRAY CHARACTERISTICS / *VORTEX GENERATORS / *VORTEX SHEETS

Reference no 69

82A25773 NASA ISSUE 11 Category 8

Sensitivity of helicopter aeromechanical stability to dynamic inflow

GAONKAR, G. H.; MITRA, A. K.; REDDY, T. S. R.; PETERS, D. A.

820000 NSF CME-79-06304 Vertica, vol. 6, no. 1, 1982, p. 59-75. AC(Indian Institute of Science, Bangalore, India) AD(Washington University, St. Louis, MO) 17 p. Refs. 14 Jpn. 1708

Since aeromechanical instability or 'resonance' of nonarticulated helicopters generally involves low-frequency flap and lead-lag regressing modes as well as a coupled rotor-body pitch or roll mode, it is expected to be appreciably influenced by dynamic inflow. Therefore, a resonance analysis in hover is presented with four objectives concerning (1) the effect of dynamic inflow on air and ground resonance, (2) the adequacy of the equivalent Lock number and drag coefficient to account for the effect, (3) the possible improvement in correlation with recent experimental data, with the inclusion of inflow and (4) a better understanding of the effect of rotor parameters on the resonance phenomena with dynamic inflow. Dynamic inflow is found to appreciably decrease the damping of body modes, although it increases the damping of the lag regressing mode. The effect can be approximately treated in most cases by the much simpler method of equivalent Lock number and drag coefficient. C.D.

Controlled Terms: *AERODYNAMIC COEFFICIENTS / *AERODYNAMIC DRAG / *AERODYNAMIC STABILITY / AIRCRAFT STABILITY / COMPUTATIONAL FLUID DYNAMICS / DAMPING / DEGREES OF FREEDOM / *HELICOPTER CONTROL / *HOVERING / NUMERICAL ANALYSIS / *ROTOR AERODYNAMICS / STIFFNESS

Reference no 70

82A23316 NASA ISSUE 9 Category 3

Utilizing the helicopter's versatility to improve the ATC system

DELUCIEN, A. G. B.; SMITH, F. D.

800000 In: Air Traffic Control Association, Annual Fall Conference, 25th, Arlington, VA, October 19-24, 1980, Proceedings. (A82-23309 09-04) Arlington, VA, Air Traffic Control Association, 1980, p. 89-96. AB(Pacer Systems, Inc., Arlington, VA) 8 p. Refs. 14 Jpn. 1338

This paper presents an overview of certain operating environments of civilian helicopters, and summarizes selected performance-maneuver envelopes of those models certified for operation under instrument flight rules (IFR). It discusses how the capabilities and limitations which are unique to these vehicles can be used to improve the capability, and expand the capacity, of the Air Traffic Control (ATC) system. Helicopter-oriented technical issues which will impact future changes to the system are identified. The paper concludes that as controllers become increasingly aware of related helicopter characteristics and operating environments, and take advantage of them, the effect would be to reduce controller workload; and to increase the capacity, expand the functional capabilities, and enhance the overall versatility of the ATC system. (Author)

Controlled Terms: *AIR TRAFFIC CONTROL / *AIRCRAFT MANEUVERS / AIRSPACE / AIRSPEED / CERTIFICATION / FLIGHT CHARACTERISTICS / *HELICOPTER PERFORMANCE / *INSTRUMENT FLIGHT RULES / *WORKLOADS (PSYCHOPHYSIOLOGY)

Reference no 71

82N23216# NASA ISSUE 14 Category 3

An assessment of various side-stick controller/stability and control augmentation systems for night nap-of-Earth flight using piloted simulation

LANDIS, K. H.; AIKEN, E. W.
Boeing Vertol Co., Philadelphia, Pa. (BR870123)
820400 In NASA. Ames Research Center Helicopter handling Qualities p 75-96 (SEE N82-23208 14-03) AB(Army Research and Technology Labs., Moffett Field, Calif.) 22 p. Jpn. 1899 HC A11/MF A01

Several night nap-of-the-earth mission tasks were evaluated using a helmet-mounted display which provided a limited field-of-view image with superimposed flight control symbology. A wide range of stability and control augmentation designs was investigated. Variations in controller force-deflection characteristics and the number of axes controlled through an integrated side-stick controller were studied. In general, a small displacement controller is preferred over a stiffstick controller particularly for maneuvering flight. Higher levels of stability augmentation were required for IMC tasks to provide handling qualities comparable to those achieved for the same tasks conducted under simulated visual flight conditions. T.M.

Controlled Terms: AIRCRAFT MANEUVERS / *CONTROLLABILITY / *CONTROLLERS / *FLIGHT CONTROL / FLIGHT SIMULATION / *HELICOPTER CONTROL / HELMET MOUNTED DISPLAYS / MILITARY HELICOPTERS / *NAP-OF-THE-EARTH NAVIGATION / *NIGHT FLIGHTS (AIRCRAFT) / NIGHT VISION / PILOT PERFORMANCE / *STABILITY AUGMENTATION / VISUAL FLIGHT

Reference no 72
82N23215# NASA ISSUE 14 Category 3
Unified results of several analytical and experimental studies of helicopter handling qualities in visual terrain flight
CHEN, R. T. N.
National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)
820400 In its Helicopter Handling Qualities p 59-74 (SEE N82-23208 14-03) 16 p. Jpn. 1899 HC A11/MF A01

The studies were undertaken to investigate the effects of rotor design parameters, interaxis coupling, and various levels of stability and control augmentation on the flying qualities of helicopters performing low-level, terrain-flying tasks in visual meteorological conditions. Some unified results are presented, and the validity and limitations of the flying-qualities data obtained are interpreted. Selected results, related to various design parameters, provide guidelines for the preliminary design of rotor systems and aircraft augmentation systems. T.M.

Controlled Terms: *AIRCRAFT MANEUVERS / AIRCRAFT SPECIFICATIONS / *CONTROLLABILITY / FLIGHT CHARACTERISTICS / HELICOPTER CONTROL / *HELICOPTER PERFORMANCE / METEOROLOGICAL PARAMETERS / *NAP-OF-THE-EARTH NAVIGATION / *ROTARY WINGS / *STABILITY AUGMENTATION / TERRAIN FOLLOWING AIRCRAFT / *VISUAL FLIGHT / WEATHER

Reference no 73
82N23214# NASA ISSUE 14 Category 3
A helicopter handling-qualities study of the effects of engine response characteristics, height-control dynamics, and excess power on nap-of-the-Earth operations
CORLISS, L. D.
Army Research and Technology Labs., Moffett Field, Calif. (A2025071)
820400 In NASA. Ames Research Center Helicopter Handling Qualities p 47-57 (SEE N82-23208 14-03) 11 p. Jpn. 1899 HC A11/MF A01

The helicopter configuration with an rpm-governed gas-turbine engine was examined. A wide range of engine response time, vehicle damping and sensitivity, and excess power levels was studied. The data are compared with the existing handling-qualities specifications, MIL-F-83300 and AGARD 577, and in general show a need for higher minimums when performing such NOE maneuvers as a dolphin and bob-up task. T.M.

Controlled Terms: *AIRCRAFT MANEUVERS / *CONTROLLABILITY / *DYNAMIC RESPONSE / ELECTRIC CONTROL / *ENGINE CONTROL / FLIGHT SIMULATORS / FUEL CONTROL / FUEL INJECTION / *GAS TURBINE ENGINES / *HELICOPTER CONTROL / *HELICOPTER ENGINES / *NAP-OF-THE-EARTH NAVIGATION

Reference no 74
82N23213# NASA ISSUE 14 Category 3
Flight tests for the assessment of task performance and control activity
PAUSDER, H. J.; HUMMES, D.
Deutsche Forschungs- und Versuchsanstalt fuer Luft- und Raumfahrt, Brunswick (West Germany). (D0696666)
820400 Inst. fuer Flugmechanik. In NASA. Ames Research Center Helicopter Handling Qualities p 35-46 (SEE N82-23208 14-03) 12 p. Jpn. 1899 HC A11/MF A01

The tests were performed with the helicopters BO 105 and UH-1D. Closely connected with tactical demands the six test pilots' task was to minimize the time and the altitude over the obstacles. The data reduction yields statistical evaluation parameters describing the control activity of the pilots and the achieved task performance. The results are shown in form of evaluation diagrams. Additionally dolphin tests with varied control strategy were performed to get more insight into the influence of control techniques. From these test results recommendations can be derived to emphasize the direct force control and to reduce the collective to pitch crosscoupling for the dolphin. T.M.

Controlled Terms: *AIRCRAFT MANEUVERS / ATTITUDE CONTROL / *BO 105 HELICOPTER / CONTROLLABILITY / DATA ACQUISITION / *FLIGHT CONTROL / *FLIGHT TESTS / *HELICOPTER CONTROL / HELICOPTER PERFORMANCE / HOVERING / MILITARY OPERATIONS / *NAP-OF-THE EARTH NAVIGATION / *PILOT PERFORMANCE / *UH 1 HELICOPTER

Reference no 75
82N23212# NASA ISSUE 14 Category 3
Influence of maneuverability on helicopter combat effectiveness
FALCO, M.; SMITH, R.
Grumman Aerospace Corp., Bethpage, N.Y. (G6919090)
820400 In NASA. Ames Research Center Helicopter Handling Qualities p 23-33 (SEE N82-23208 14-03) AB(Army Aviation Research and Development Command, St. Louis, Mo.) 11 p. Jpn. 1899 HC A11/MF A01

interrelationship between stability and controllability. The application potential and experience of PI methods for dynamic wind tunnel testing and requirements necessary for gaining more insight and confidence in using static and dynamic wind tunnel data for flight/ground testing corelation are discussed. E.A.K.

Controlled Terms: *AIRCRAFT STABILITY / *DYNAMIC STABILITY / *FIXED WINGS / INDEPENDENT VARIABLES / MATHEMATICAL MODELS / *PARAMETER IDENTIFICATION / *ROTARY WINGS / *STABILITY DERIVATIVES / STATISTICAL ANALYSIS / SYSTEMS ENGINEERING / *WIND TUNNEL TESTS

Reference no 125

81N30140# NASA ISSUE 21 Category 8

Parameter identification of a hingeless rotor helicopter in flight conditions with increased instability

KLOSTER, M.; KALETKA, J.; SCHAEUFELE, H.

Messerschmitt-Boelkow-Blohm G.m.b.H., Ottobrunn (West Germany). (MT620643)

MBB-UD-307/80-OE 800901 Unternehmensbereich Drehfluegler. Presented at 6th European Rotocraft and Powered Lift Aircraft Forum, Bristol, England, 16-19 Sep. 1980 AB(DFVLR, Brunswick) 24 p. Jpn. 2871 HC A02/MF A01

Hover and flight at maximum speed were studied. Both flight conditions were performed with maximum weight and a mid center of gravity position. An attitude feedback control system was necessary because of the instability. A strapdown system was used for measurements. Closed loop stabilization was obtained with an onboard computer. The input signals of the unstabilized helicopter were optimized. Time and frequency calculations show that special input signals are needed for the closed loop system. A special distribution of the power spectrum leads to a quasi-optimized input signal. These signals were filtered with second order linear filters to suppress rotor dynamics, derivatives identified from flight tests (six degree of freedom rigid body model) are compared with the results of nonlinear simulation and quasistatic theory. In general, agreement is good. Author (ESA)

Controlled Terms: AIRBORNE/SPACEBORNE COMPUTERS / *AIRCRAFT STABILITY / ATTITUDE CONTROL / FEEDBACK CONTROL / FLIGHT SIMULATION / *HELICOPTER CONTROL / *HOVERING STABILITY / INDEPENDENT VARIABLES / LINEAR FILTERS / *PARAMETER IDENTIFICATION / *ROTARY WINGS / *STABILITY DERIVATIVES

Reference no 126

81A29172 NASA ISSUE 12 Category 8

The effect of the atmospheric turbulence on the rotor blade flap-leadlag motion stability in hovering

FUJIMORI, Y.

AIAA 81-0610 810000 In: Structures, Structural Dynamics and Materials Conference, 22nd, Atlanta, Ga., April 6-8, 1981, and AIAA Dynamics Specialists Conference, Atlanta, Ga., April 9, 10, 1981, Technical Papers. Part 2. (A81-29428 12-01) New York, American Institute of Aeronautics and Astronautics, Inc., 1981, p. 406-415. AA(National Aerospace Laboratory, Tokyo, Japan) 10 p. Refs. 7

The general procedure has been proposed to study the stochastic stability characteristics of nonlinear differential equations with random parametric excitations. The method is based on the existence of the equilibrium of the deterministic system and the parametric excitations being infinitesimally small. The method has been applied to evaluate the motion stability of the coupled flap-leadlag motion in three dimensional atmospheric turbulence. The numerical examples of zero advance ratio flight show that the first and second moment stabilities possess almost identical boundaries and that the atmospheric turbulence has a favorable effect on the system stability. (Author)

Controlled Terms: *AERODYNAMIC STABILITY / *ATMOSPHERIC TURBULENCE / EQUILIBRIUM EQUATIONS / FLAPS (CONTROL SURFACES) / *HOVERING STABILITY / NONLINEAR EQUATIONS / *ROTOR AERODYNAMICS / *ROTOR BLADES (TURBOMACHINERY) / STOCHASTIC PROCESSES

Reference no 127

81N29118 NASA ISSUE 20 Category 5

AH-1S(Prod) airworthiness and flight characteristics for instrument flight

Final Report, May - Aug. 1980

TULLOCH, J. S.; OTTOMEYER, J. D.; FRANKENBERGER, C. E., JR.; PICASSO, B. D., III

Army Aviation Engineering Flight Activity, Edwards AFB, Calif. (AZ102498)

AD-A100946; USAAEFA-79-08 801100 47 p. Jpn. 2725 HC A03/MF A01

The United States Army Aviation Engineering Flight Activity conducted an Airworthiness and Instrument Flight Characteristics evaluation of a Production AH-1S (Prod) to determine potential for the AH-1S with Enhanced Cobra Armament System (ECAS) to meet instrument meteorological conditions qualification criteria. The test aircraft was configured with two tube launched, optically tracked, wireguided (TOW) missile launchers on each outboard wing stores station and a 7-tube lightweight launcher on each inboard wing stores station. The test consisted of 16.3 flight hours which were flown during 12 test flights. Four deficiencies and seven shortcomings associated with flying the AH-1S in instrument flight conditions, were identified. The deficiencies identified were: (1) unsatisfactory cyclic control system mechanical characteristics; (2) large pitot-static system airspeed errors in climb and descent; (3) easily excited lateral gust response; (4) vertigo-inducing location of radio control panels. Five specification noncompliances were noted. The AH-1S (Prod) is not suitable for flight in instrument meteorological conditions, which infers that the AH-1S (ECAS) will also not be suitable. Author (GRA)

Controlled Terms: *AERODYNAMIC STABILITY / *AIRCRAFT RELIABILITY / DYNAMIC RESPONSE / EXTERNAL STORES / *FLIGHT CONTROL / FLIGHT INSTRUMENTS / *HELICOPTERS / *TOW MISSILES

Reference no 121

81N33148# NASA ISSUE 24 Category 5

The impact of helicopter flight mechanics on mission performance

GMELIN, B. L.; PAUSDER, H. J.

Institut fuer Flugmechanik, Brunswick (West Germany). (IH545557)

810600 In AGARD The Impact of Mil. Appl. on Rotorcraft and V/STOL Aircraft Design 14 p (SEE N81-33137 24-01) 14 p. Jpn. 3288 HC A12/MF A01

In comparison to other VTOL aircraft, the helicopter has inherently superior flying characteristics at hover and in the low speed region but it has clear disadvantages by its lower high speed capability. For the completion of today's military helicopter missions near the ground and in all weather flight conditions a careful flight mechanical adaptation of the pilot-helicopter system to the specific task is required. This includes the combined optimization of the basic aircraft, the additional system equipment, and in a certain degree the pilot control behavior. A DFVLR-methodology is presented, having the objective to evaluate the flight mechanical characteristics of the pilot-helicopter system with regard to specific flight tasks. The method leads to taskoriented evaluation diagrams which include scales for the overall system task performance and for the demands on the pilot. Relevant flight test results of the BO 105 helicopter in hover and NOE-flight are presented. Especially the parameters control activity and task performance are discussed. Author

Controlled Terms: AIRCRAFT DESIGN / AIRCRAFT PERFORMANCE / *BO-105 HELICOPTER / *FLIGHT CHARACTERISTICS / FLIGHT CONTROL / *FLIGHT MECHANICS / *HELICOPTER DESIGN / *HELICOPTER PERFORMANCE / *HELICOPTERS / V/STOL AIRCRAFT / VERTICAL TAKEOFF AIRCRAFT

Reference no 122

81A32010 NASA ISSUE 13 Category 5

Use of multiblade sensors for on-line rotor tip-path plane estimation

DU VAL, R. W.

National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)

801000 American Helicopter Society, Journal, vol. 25, Oct. 1980, p. 13-21. AA(NASA, Ames Research Center, Helicopter Technology Div., Moffett Field, Calif.) 9 p. Refs. 5 Jpn. 2107

Techniques are investigated for on-line estimation of rotor states in the nonrotating frame from multiple, simultaneous measurements in the rotating frame. The multiblade coordinate transformation is first applied to transform both flapping and flapping rate measurements into the nonrotating frame. The 'observer' approach is then used to generate algorithms for estimating tip-path plane rate and attitude from transformed flapping and flapping rate measurements. A numerical evaluation using simulated measurements is conducted to evaluate the performance of the algorithms and recommendations are made. (Author)

Controlled Terms: BLADE TIPS / COMPUTERIZED SIMULATION / ERROR ANALYSIS / FILTRATION / *FLAPPING / *HELICOPTER CONTROL / MATRICES (MATHEMATICS) / NUMERICAL ANALYSIS / PITCH (INCLINATION) / *ROTARY WINGS / *ROTOR AERODYNAMICS / *ROTOR BLADES / SENSORS / *TIP SPEED / TRANSIENT RESPONSE

Reference no 123

81N31232# NASA ISSUE 22 Category 9

The effects of various fidelity factors on simulated helicopter hover

Final Report, Mar. 1979 - Jun. 1980

RICARD, G. L.; PARRISH, R. V.; ASHWORTH, B. R.; WELLS, M. D.

Naval Training Equipment Center, Orlando, Fla. (NT855045)

AD-A102028; NAVTRAEQUIPC-IH-321 810100 72 p. Jpn. 3021 HC A04/MF A01

The effects of the cues of aircraft motion, of delays in a visual scene, and of movement of a ship model were examined by measuring pilots' ability to hover a simulated helicopter near a destroyer class ship. Fourteen Navy helicopter pilots were tested in a within subjects, factorial combination of fixed base, moving base, and g-seat conditions where delays of approximately 66 or 128 milliseconds existed in the simulator's visual display, and the pilots were to hover near a moving or stationary ship. In addition, an effort was made to determine the effect a head-up display of aircraft position had on the measures of control. Best performance was seen with the moving base simulation while poorest control was associated with the fixed-base conditions and in-between performance was measured under the g-seat conditions. The addition of the longer delay uniformly elevated scores, but movement of the ship model had little effect. Also performance was not affected by removal of the head-up display. A recommendation is made for the configuration of trainers for aircrews of marginally stable vehicles. This is that motion cuing is likely to be useful for flight regimes such as hover, and that currently platform technology is the recommended source of these cues. Author (GRA)

Controlled Terms: AIRCRAFT CARRIERS / AIRCRAFT LANDING / *FLIGHT SIMULATORS / *HEAD-UP DISPLAYS / *HELICOPTER CONTROL / *HOVERING / *MAN MACHINE SYSTEMS / MOTION SIMULATORS / PILOT PERFORMANCE / TRAINING SIMULATORS / VISUAL PERCEPTION

Reference no 124

81N31115# NASA ISSUE 22 Category 5

Determination of aircraft dynamic stability and control parameters from flight testing

HAMEL, P. G.

Deutsche Forschungs- und Versuchsanstalt fuer Luft- und Raumfahrt, Brunswick (West Germany). (D0696666)

810500 Inst. fuer Flugmechanik. In AGARD Dyn. Stability Parameters 42 p (SEE N81-31105 22-01) 42 p. Jpn. 3004 HC A17/MF A01

The present state of the art of aircraft parameter identification (PI) techniques from flight test data and appraisal of current methods developed and applied to various aircraft configurations and flight conditions are reviewed. Practical aspects and results of PI techniques are emphasized. This is especially relevant for data correlation and for increasing confidence in static and dynamic wind tunnel prediction techniques. Recent experience for fixed and rotary wing aircraft PI are presented as well as identification results for extreme flight regimes. Information on pilot in the loop and closed loop aspects of aircraft PI are given with special reference to the

second-order integration scheme. It is found that the present ordering scheme does not achieve completely equivalent models based on mixed and displacement formulations, due to the neglect of the order of squares of rotation with respect to unity. Numerical results for a hypothetical nonuniform blade, including the nonlinear static equilibrium solution, were obtained with effort and computer time equivalent to that required for a uniform blade. O.C.

Controlled Terms: *AERODYNAMIC LOADS / COLLOCATION / COMPUTER PROGRAMS / DIFFERENTIAL EQUATIONS / EIGENVALUES / EQUATIONS OF MOTION / *HOVERING STABILITY / LINEAR EQUATIONS / *MATHEMATICAL MODELS / NONLINEAR EQUATIONS / NONUNIFORMITY / PERTURBATION THEORY / *ROTARY WINGS / ROTOR AERODYNAMICS / *STRUCTURAL STABILITY

Reference no 117
81A39900 NASA ISSUE 18 Category 8
Gust response of rotary wing aircraft and its alleviation
SAITO, S.; AZUMA, A.; NAGAO, M.
810000 Vertica, vol. 5, no. 2, 1981, p. 173-184. AB(Tokyo, University, Tokyo, Japan) AC(Olympus Optical Co., Ltd., Tokyo, Japan) 12 p. Refs. 12

A simple feedback mechanism is described for helicopter rotor gust response alleviation, comprising sensors for the detection of flapping motion in rotor blades and in pitch-control actuators which are controlled by gust-alleviation signals. The validity of the system is demonstrated by applying a theoretical calculation, based on local momentum theory, to the responses of a helicopter that is considered as a complete dynamic system while it penetrates into (1) a sinusoidal gust and (2) a step gust. Extensive dimensional data are furnished for both the rotor and the helicopter exemplified, as well as thrust coefficient comparisons for the feedback and non-feedback systems. O.C.

Controlled Terms: *FEEDBACK CONTROL / FLAPPING / *GUST ALLEVIATORS / *GUST LOADS / *HELICOPTER CONTROL / *ROTARY WINGS / *ROTOR AERODYNAMICS / ROTOR BLADES

Reference no 118
81A39899 NASA ISSUE 18 Category 2
Co-axial rotor aerodynamics in hover
ANDREW, M. J.
810000 (European Rotorcraft and Powered Lift Aircraft Forum, 6th, Bristol, England, Sept. 16-19, 1980.) Vertica, vol. 5, no. 2, 1981, p. 163-172. Research supported by the Science Research Council. AA(Southampton, University, Southampton, England) 10 p. Refs. 32

A remotely piloted, coaxial contrarotating twin rotor (CCTR) helicopter extensively modified for research was used to investigate the hover aerodynamics of such configurations. Good agreement has been found between experimental, induced downwash distributions and overall rotor performances and a theoretical model based on momentum, blade element and vortex theories. Semiempirical equations are also derived for the initial viscous vortex core size and maximum swirl velocities. It is shown that the conventional comparison of the CCTR with one of its own rotors is false, in that the single rotor is thrust-limited by the onset of blade stall. When compared with a single rotor having the same thrust potential, the theory presented indicates that the CCTR layout generates more thrust per unit power in hover through a reduction in induced power of approximately 5%. O.C.

Controlled Terms: *AIRCRAFT MODELS / AIRCRAFT WAKES / COMPUTERIZED SIMULATION / FLOW EQUATIONS / *HELICOPTER DESIGN / *HOVERING STABILITY / *ROTOR AERODYNAMICS / ROTOR BLADES / TRAILING EDGES / *VORTICITY EQUATIONS / *WING TIP VORTICES

Reference no 119
81A39898 NASA ISSUE 18 Category 5
Experimental and analytical studies of a model helicopter rotor in hover
CARADONNA, F. X.; TUNG, C.
810000 Vertica, vol. 5, no. 2, 1981, p. 149-161. AB(U. S. Army, Aeromechanics Laboratory, Moffett Field, CA) 13 p. Refs. 18

The present study is a benchmark test to aid the development of various rotor performance codes. The study involves simultaneous blade pressure measurements and tip vortex surveys. Measurements were made for a wide range of tip Mach numbers including the transonic flow regime. The measured tip vortex strength and geometry permit effective blade loading predictions when used as input to a prescribed wake lifting surface code. It is also shown that with proper inflow and boundary layer modeling, the supercritical flow regime can be accurately predicted. (Author)

Controlled Terms: BLADE TIPS / *GROUND TESTS / *HELICOPTERS / HOT-WIRE FLOWMETERS / *HOVERING STABILITY / LIFT / LOAD DISTRIBUTION (FORCES) / MACH NUMBER / *PERFORMANCE PREDICTION / PITCH (INCLINATION) / *PRESSURE DISTRIBUTION / *ROTARY WINGS / SUPERCRITICAL FLOW / TEST CHAMBERS / TRANSONIC FLOW / *VORTICES / WAKES

Reference no 120
81N33194# NASA ISSUE 24 Category 5
Design of a slot height distribution for increased hover control power on a circulation control rotor
Final Report
POE, D. W.
Naval Ship Research and Development Center, Bethesda, Md. (NT691395)
AD-A103535; DTNSRDC/ASED-80/24 801200 Aviation and Surface Effects Dept. 31 p. Jpn. 3296 HC A03/MF A01

The Circulation Control Rotor Performance Prediction computer program was used with the XH-2/CCR rotor configuration to determine a slot height distribution that would improve control power in hover without causing excessive cyclic pressure requirements for trim in forward flight. Effects of total slot area as well as distribution were considered. The final distribution was constrained by a minimum practical slot height setting of 0.002 in. and a minimum unpressurized blade slot area of 3.0 square inches. Several distributions were evaluated. Noteworthy trends that emerged are: (1) A negatively tapered slot height distribution is favorable to producing hub moments in hover, and (2) a uniform distribution (zero taper) requires the lowest cyclic pressure for trim at 120 knots. The final distribution selection exhibited a 38-percent improvement in predicted hub moment over a slot height distribution previously used on the flight demonstrator. Author (GRA)

Controlled Terms: AERODYNAMIC STABILITY / *HOVERING STABILITY / JET FLOW / *PROPELLER BLADES / RETROFITTING / *ROTARY WINGS / *SLOTS / TRAILING EDGES

trend for all of the aircraft considered. The low speed forward flight data yield a single non-dimensional curve that relates the power required at any given speed to that required at hover and at the minimum power speed. The vertical climb data also yield a single non-dimensional plot that relates the climb power requirement to the climb rate. For forward climb, a single climb constant was found that is valid for all climb rates. (Author)

Controlled Terms: *CLIMBING FLIGHT / COMPUTER TECHNIQUES / DATA REDUCTION / FLIGHT CHARACTERISTICS / *HELICOPTER PERFORMANCE / *HOVERING STABILITY / *LOW SPEED STABILITY / *PERFORMANCE PREDICTION

Reference no 113

81A40114 NASA ISSUE 18 Category 8

Lateral flutter of loads towed beneath helicopters and its avoidance

SIMPSON, A.; FLOWER, J. W.

800000 In: European Rotorcraft and Powered Lift Aircraft Forum, 6th, Bristol, England, September 16-19, 1980, Conference Papers. Part 2. (A81-40076 18-01) Bristol, University of Bristol, 1980. 23 p. AB(Bristol, University, Bristol, England) 23 p. Refs. 9

The aerodynamic instability problem of strop-supported freight loads is addressed. An in-depth study is made of the lateral, low-frequency flutter of rectangular cargo containers supported by multi-strop arrangements, and it is shown that necessary conditions for flutter and divergence may be obtained in which the primary parameters are the static strop tensions at the current forward speed. These criteria allow rapid assessment of lateral stability when only the longitudinal static aerodynamic characteristics are available for analysis. An overview is also given of the state of the art of steady and unsteady aerodynamics for the case of towed bodies, and the need for more fundamental research in this area is highlighted. O.C.

Controlled Terms: *AERODYNAMIC STABILITY / CARGO / COMPUTER TECHNIQUES / EQUATIONS OF MOTION / *FLUTTER / *HELICOPTER CONTROL / *LATERAL STABILITY / QUASI-STEADY STATES / *TOWED BODIES / WIND TUNNEL TESTS

Reference no 114

81A40098 NASA ISSUE 18 Category 2

Aerodynamic study of a hovering rotor

POURADIER, J. M.; HOROWITZ, E.

800000 In: European Rotorcraft and Powered Lift Aircraft Forum, 6th, Bristol, England, September 16-19, 1980, Conference Papers. Part 1. (A81-40076 18-01) Bristol, University of Bristol, 1980. 14 p. AA(Societe Nationale Industrielle Aerospatiale, Division Helicopteres, Marignane, Bouches-du-Rhone, France) AB(ONERA, Chatillon-sous-Bagneux, Hauts-de-Seine, France) 14 p. Refs. 14

Part of a joint research program between ONERA and Aerospatiale, aimed to improve helicopter rotor performance, emphasizes an aerodynamic study of hovering flight. A wind tunnel test was performed on a three-bladed rotor, using a two-component laser velocimeter, to determine the tip vortex path and to study tip vortex core and development. The tip vortex path was determined in two vertical planes - the longitudinal axis and the transverse axis of the airflow vein, and except for three velocity peaks induced tangential velocities proved to be four or five times lower than axial velocities. Success of the experiment demonstrated that the laser velocimeter can be used to study wake geometry and the formation and dilution of various vortex components. A theoretical method for performance calculation is then presented, which is based on the vortex theory and provides for partial free wake analysis. Calculations with free wake analysis closely approximated experimental results, whereas prescribed wake calculations often showed erroneous trends. Moreover, calculated and measured tip vortex paths were accurately predicted, whereas the prescribed wake analysis gave too high an axial velocity. J.F.

Controlled Terms: AIR FLOW / BLADE TIPS / FLOW VELOCITY / *HELICOPTER PERFORMANCE / *HELICOPTER WAKES / *HOVERING STABILITY / LASER DOPPLER VELOCIMETERS / RADIAL DISTRIBUTION / *ROTARY WINGS / *ROTOR AERODYNAMICS / VELOCITY DISTRIBUTION / VORTICES / *WIND TUNNEL TESTS

Reference no 115

81A40097 NASA ISSUE 18 Category 2

Experimental and analytical studies of a model helicopter rotor in hover

CARADONNA, F. X.; TUNG, C.

800000 In: European Rotorcraft and Powered Lift Aircraft Forum, 6th, Bristol, England, September 16-19, 1980, Conference Papers. Part 1. (A81-40076 18-01) Bristol, University of Bristol, 1980. 20 p. AB(U.S. Army, Aeromechanics Laboratory, Moffett Field, CA) 20 p. Refs. 18

The present study is a benchmark test to aid the development of various rotor performance codes. The study involves simultaneous blade pressure measurements and tip vortex surveys. Measurements were made for a wide range of tip Mach numbers including the transonic flow regime. The measured tip vortex strength and geometry permit effective blade loading predictions when used as input to a prescribed wake lifting surface code. It is also shown that with proper inflow and boundary layer modeling, the supercritical flow regime can be accurately predicted. (Author)

Controlled Terms: *HELICOPTER PERFORMANCE / HOT-WIRE ANEMOMETERS / *HOVERING / MACH NUMBER / PRESSURE DISTRIBUTION / *ROTARY WINGS / *ROTOR AERODYNAMICS / TRANSONIC FLOW / WAKES / WING TIP VORTICES

Reference no 116

81A40088 NASA ISSUE 18 Category 5

Stability of nonuniform rotor blades in hover using a mixed formulation

STEPHENS, W. B.; HODGES, D. H.; AVILA, J. H.; KUNG, R.-M.

800000 In: European Rotorcraft and Powered Lift Aircraft Forum, 6th, Bristol, England, September 16-19, 1980, Conference Papers. Part 1. (A81-40076 18-01) Bristol, University of Bristol, 1980. 20 p. AB(U.S. Army, Aeromechanics Laboratory, Moffett Field, CA) AD(Technology Development of California, Santa Clara, CA) 20 p. Refs. 24

A mixed formulation for calculating the static equilibrium and stability eigenvalues of nonuniform rotor blades in hover is presented, in which the static equilibrium equations are linear and solved by an accurate and efficient collocation method and the linearized perturbation equations are solved by a one-step,

Controlled Terms: *AERODYNAMIC STABILITY / COMPUTERIZED SIMULATION / FEEDBACK CONTROL / *FLIGHT CONDITIONS / *FLIGHT TESTS / *HELICOPTER PERFORMANCE / LATERAL CONTROL / LONGITUDINAL CONTROL / OPTIMIZATION / *PARAMETER IDENTIFICATION / RIGID ROTORS / *SIGNAL PROCESSING / YAWING MOMENTS

Reference no 109

81A40122 NASA ISSUE 18 Category 8

On the use of approximate models in helicopter flight mechanics

PADFIELD, G. D.

800000 In: European Rotorcraft and Powered Lift Aircraft Forum, 6th, Bristol, England, September 16-19, 1980, Conference Papers. Part 2. (A81-40076 18-01) Bristol, University of Bristol, 1980. 24 p. AA(Royal Aircraft Establishment, Bedford, England) 24 p. Refs. 11

This paper addresses several aspects of the prediction of helicopter flight behavior and emphasises the need for low order approximations to aid physical interpretation of important flying qualities. The centre spring, rigid blade rotor model is used for predicting the integrated loads from hingeless and articulated rotors. Stability derivatives, derived with this model, are then used in the search for simplified approximations to the short term pitch attitude response to cyclic pitch control, throughout the speed range. The method of weakly coupled systems provides a mathematical framework for the analysis which is applied to the prediction of flight path trajectories during transient manoeuvres. The use of truncated dynamic models for combined pitch and roll maneuvres is also discussed. (Author)

Controlled Terms: *AERODYNAMIC STABILITY / *AIRCRAFT MODELS / APPROXIMATION / *DYNAMIC MODELS / *FLIGHT MECHANICS / *FLIGHT SIMULATION / *HELICOPTER CONTROL / *HELICOPTER PERFORMANCE / LONGITUDINAL STABILITY / MATHEMATICAL MODELS / NUMERICAL ANALYSIS / *PERFORMANCE PREDICTION / PITCH (INCLINATION) / ROLL

Reference no 110

81A40121 NASA ISSUE 18 Category 5

Flight tests and statistical data analysis for flying qualities investigations

SANDERS, K.; PAUSDER, H.-J.; HUMMES, D.

800000 In: European Rotorcraft and Powered Lift Aircraft Forum, 6th, Bristol, England, September 16-19, 1980, Conference Papers. Part 2. (A81-40076 18-01) Bristol, University of Bristol, 1980. 25 p. AC(Deutsche Forschungs- und Versuchsanstalt fuer Luft- und Raumfahrt, Institut fuer Flugmechanik, Braunschweig, West Germany) 25 p. Refs. 5

A new test and analysis technique has been developed and proven as a valuable tool for the evaluation of helicopter closed loop flying qualities in mission-oriented environments. A detailed description is presented of the application of the technique to a hovering tracking mission element, selected from German anti tank helicopter mission specifications. The statistical evaluation of the hovering tracking tests contains the determination of such parameters as standard deviations, variances, and peak-to-peak values. The dynamical content of the measuring signals can be represented in such terms as steady states, reversals and continuous movements. Diagrams are presented in which boundaries for pilot stress rating groups point out the optimal combination of tracking effectiveness and control activity. O.C.

Controlled Terms: DATA REDUCTION / FEEDBACK CONTROL / *FLIGHT CHARACTERISTICS / *FLIGHT TESTS / *HELICOPTER CONTROL / *HELICOPTER PERFORMANCE / *HOVERING STABILITY / *LOW SPEED STABILITY / MEAN SQUARE VALUES / OBSTACLE AVOIDANCE / PROBABILITY DENSITY FUNCTIONS / STATISTICAL ANALYSIS

Reference no 111

81A40120 NASA ISSUE 18 Category 8

A pilot's assessment of helicopter handling-quality factors common to both agility and instrument flying tasks

GERDES, R. M.

National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)

800000 In: European Rotorcraft and Powered Lift Aircraft Forum, 6th, Bristol, England, September 16-19, 1980, Conference Papers. Part 2. (A81-40076 18-01) Bristol, University of Bristol, 1980. 18 p. AA(NASA, Ames Research Center, Moffett Field, CA) 18 p. Refs. 7

Results from a series of simulation and flight investigations undertaken to evaluate helicopter flying qualities and the effects of control system augmentation for nap-of-the-earth (NOE) agility and instrument flying tasks were analyzed to assess handling-quality factors common to both tasks. Precise attitude control was determined to be a key requirement for successful accomplishment of both tasks. Factors that degraded attitude controllability were improper levels of control sensitivity and damping and rotor-system cross-coupling due to helicopter angular rate and collective pitch input. Application of rate-command, attitude-command, and control-input decouple augmentation schemes enhanced attitude control and significantly improved handling qualities for both tasks. NOE agility and instrument flying handling-quality considerations, pilot rating philosophy, and supplemental flight evaluations are also discussed. (Author)

Controlled Terms: AERODYNAMIC STABILITY / ATTITUDE CONTROL / *CONTROLLABILITY / FLIGHT CHARACTERISTICS / *FLIGHT CONTROL / FLIGHT SIMULATION / *HELICOPTER CONTROL / HELICOPTER PERFORMANCE / HELICOPTER PROPELLER DRIVE / INSTRUMENT FLIGHT RULES / MANEUVERABILITY / *MANUAL CONTROL / MILITARY HELICOPTERS / *NAP-OF-THE-EARTH NAVIGATION / *PILOT PERFORMANCE / ROTARY WINGS

Reference no 112

81A40118 NASA ISSUE 18 Category 5

An empirical prediction method for helicopter performance in low speed level flight and in vertical and forward flight climbs

SMITH, E. H.

800000 In: European Rotorcraft and Powered Lift Aircraft Forum, 6th, Bristol, England, September 16-19, 1980, Conference Papers. Part 2. (A81-40076 18-01) Bristol, University of Bristol, 1980. 15 p. AA(U.S. Navy, Naval Air Systems Command, Washington, DC) 15 p. Refs. 36

This paper presents methods for the prediction of helicopter power requirements in low speed forward flight and in vertical and forward flight climbs. The methodology is based on an extensive study of experimental data. Available analytical prediction methods are summarized and compared with the data. The data are reduced to non-dimensional forms that cause the data to collapse into a single

Controlled Terms: AERODYNAMIC FORCES / AERODYNAMIC STABILITY / *AEROELASTICITY / *AIRCRAFT STABILITY / *AIRFRAMES / BEARINGLESS ROTORS / COMPUTERIZED SIMULATION / FLOQUET THEOREM / *FLUTTER ANALYSIS / FUSELAGES / *HELICOPTERS / HOVERING / MATHEMATICAL MODELS / *RIGID ROTORS / *ROTARY WINGS / ROTOR AERODYNAMICS / VIBRATION MODE

Reference no 105

81A40138 NASA ISSUE 18 Category 5

Hover performance methodology at Bell Helicopter Textron

KOCUREK, J. D.; BERKOWITZ, L. F.; HARRIS, F. D.

AHS 80-3 800000 In: American Helicopter Society, Annual Forum, 36th, Washington, DC, May 13-15, 1980, Proceedings. (A81-40136 18-01) Washington, DC, American Helicopter Society, 1980. 48 p. AC(Bell Helicopter Textron, Fort Worth, TX) 48 p. Refs. 17

During the past year a new hover performance methodology has become the standard for production design use at an American aerospace company. The methodology is centered around a lifting surface analysis of the isolated main and tail rotors. The analysis features an improved calculation of induced torque, a refined treatment of three-dimensional compressibility, and a circulation coupled prescribed wake that permits application to a wide variety of rotors. With this basic methodology, total aircraft performance is built-up by considering airframe downloading, anti-torque and accessory power requirements, and power transmission efficiencies. The considered Hover Performance Methodology establishes a structured, documented procedure and a data base from which further refinements can be made with continuity and clear recognition of technical position. G.R.

Controlled Terms: COMPRESSIBILITY EFFECTS / *HELICOPTER PERFORMANCE / *HOVERING / LIFTING BODIES / PERFORMANCE PREDICTION / *ROTARY WINGS / *ROTOR AERODYNAMICS / ROTOR LIFT / VORTICES / WAKES

Reference no 106

81A40134 NASA ISSUE 18 Category 8

Optimal higher harmonic blade pitch control for minimum vibration of a hinged rotor

BEINER, L.

800000 In: European Rotorcraft and Powered Lift Aircraft Forum, 6th, Bristol, England, September 16-19, 1980, Conference Papers. Part 2. (A81-40076 18-01) Bristol, University of Bristol, 1980. 21 p. AA(Negev, University, Beersheba, Israel) 21 p. Refs. 7

A simple rotor model is used to obtain closed form expressions for the optimal b/rev blade pitch required to suppress the b/rev hub axial force of a b-bladed hinged rotor. Explicit control laws are obtained for b = 2, 3, and 4. As a predominant influence, the required b/rev blade pitch amplitude is shown to increase with airspeed as mu to the 6th. For flight conditions and rotor characteristics in the range of usual applications, the optimal pitch amplitude does not exceed values in the range 0.04 deg for b = 4 to 1.5 deg for B = 2. The results are shown to be in satisfactory agreement with wind-tunnel test data for a 2- and 4-bladed hingeless rotor. V.L.

Controlled Terms: AERODYNAMIC CHARACTERISTICS / COEFFICIENTS / *HARMONICS / *HELICOPTER CONTROL / *HINGES / OPTIMIZATION / PERFORMANCE PREDICTION / *PITCH (INCLINATION) / *ROTARY WINGS / *ROTOR AERODYNAMICS / THRUST / *VIBRATION DAMPING / WIND TUNNEL TESTS

Reference no 107

81A40125 NASA ISSUE 18 Category 8

Identification of a linear model of rotor-fuselage dynamics from nonlinear simulation data

DUVAL, R. W.; MACKIE, D. B.

National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)

800000 In: European Rotorcraft and Powered Lift Aircraft Forum, 6th, Bristol, England, September 16-19, 1980, Conference Papers. Part 2. (A81-40076 18-01) Bristol, University of Bristol, 1980. 25 p. AB(NASA, Ames Research Center, Moffett Field, CA) 25 p. Refs. 7

Linear regression techniques are used to obtain 9- and 12-degree-of-freedom linear rotorcraft models from the input-output data generated by a nonlinear, blade-element rotorcraft simulation in hover. The resulting models are used to evaluate the coupling of the fuselage modes with the rotor flapping and lead-lag modes at various frequencies. New techniques are proposed and evaluated to improve the identification process, including a method of verifying the assumed model structure by using data sets generated at different input frequencies. (Author)

Controlled Terms: *AIRCRAFT MODELS / *AIRCRAFT STABILITY / DEGREES OF FREEDOM / *FUSELAGES / HOVERING STABILITY / LINEAR EQUATIONS / REGRESSION ANALYSIS / *ROTARY WING AIRCRAFT / *ROTOR AERODYNAMICS

Reference no 108

81A40123 NASA ISSUE 18 Category 8

Parameter identification of a hingeless rotor helicopter in flight conditions with increased instability

KLOSTER, M.; SCHAEUFELE, H.; KALETKA, J.

800000 In: European Rotorcraft and Powered Lift Aircraft Forum, 6th, Bristol, England, September 16-19, 1980, Conference Papers. Part 2. (A81-40076 18-01) Bristol, University of Bristol, 1980. 22 p. AB(Messerschmitt-Boelkow-Blohm GmbH, Munich, West Germany) AC(Deutsche Forschungs- und Versuchsanstalt fuer Luft- und Raumfahrt, Braunschweig, West Germany) 22 p. Refs. 21

Feedback control, input signal optimization and measurement equipment are described and discussed for helicopter flight test condition in which the helicopter phugoid showed increasing instability. Among the topics discussed are: (1) helicopter dynamics in hover and in forward level flight; (2) feedback control; (3) computer simulations; (4) data processiing; and (5) the identification of force, rolling moment, pitching moment, yawing moment and control derivaties. Good agreement was found between theoretical predictions and the identified derivatives. O.C.

beam combination because fewer fuselage modifications were required and retrofit was easier. In eight months of developmental flying, the Model 412, as the new configuration is designated, has accumulated over 200 flight hours, including high altitude and cold weather tests. Measurements during the preliminary flight load survey to speeds of 162 knots and maneuvers to 2.3 g indicate the design objectives have been achieved. Predicted weight and performance improvements have been verified, and the helicopter is ready for FAA certification testing. (Author)

Controlled Terms: AERODYNAMIC CHARACTERISTICS / *DYNAMIC STABILITY / ENGINE CONTROL / *ENGINE DESIGN / FLIGHT TESTS / FLUTTER / *HELICOPTER DESIGN / *HELICOPTER ENGINES / HUBS / LANDING GEAR / LOAD DISTRIBUTION (FORCES) / PERFORMANCE TESTS / *PROPULSION SYSTEM PERFORMANCE / *ROTARY WINGS / STRUCTURAL VIBRATION / VIBRATION DAMPING

Reference no 102

81A40157 NASA ISSUE 18 Category 8

X-wing stability and control development and wind tunnel demonstration tests - Helicopter, conversion, and fixed wing flight

POTTHAST, A. J.

AHS 80-27 800000 In: American Helicopter Society, Annual Forum, 36th, Washington, DC, May 13-15, 1980, Proceedings. (A81-40136 18-01) Washington, DC, American Helicopter Society, 1980. 13 p. AA(Lockheed-California Co., Burbank, CA) 13 p. Refs. 12

The X-wing aircraft combines the characteristics and attributes of a helicopter with those of a high aspect ratio transonic aircraft. A single wing/rotor lift and control system is combined with a conventional aircraft fuselage and tail. The proposed stability and control concept uses cyclic and collective modulation of rotor/wing circulation control air with hub moment feedback and conventional augmentation. The concept utilizes rotor/wing circulation control technology. A 25-foot diameter rotor/wing and breadboard control system model, for a potential X-wing demonstrator aircraft, was fabricated and tested in the NASA/Ames 40 x 80 foot wind tunnel. The tests were completed in May, 1979. A brief description of the X-wing aircraft, aerodynamic lift, and control concept which makes possible continuous lift and control during rotary, conversion between rotary and fixed, and fixed wing modes of flight is provided. Analytic methods used to predict control and stability characteristics are briefly described. Test results are compared with predictions, and conclusions are drawn relative to a control configuration for a flight aircraft. (Author)

Controlled Terms: *AIRCRAFT CONTROL / *AIRCRAFT DESIGN / *FIXED WINGS / *HELICOPTER CONTROL / HIGH ASPECT RATIO / LIFT AUGMENTATION / LONGITUDINAL CONTROL / ROTARY WINGS / SUPERSONIC AIRCRAFT / *WIND TUNNEL STABILITY TESTS / *X WING ROTORS

Reference no 103

81A40155 NASA ISSUE 18 Category 5

An experimental investigation of the effects of aeroelastic couplings on aeromechanical stability of a hingeless rotor helicopter

BOUSMAN, W. G.

AHS 80-25 800000 In: American Helicopter Society, Annual Forum, 36th, Washington, DC, May 13-15, 1980, Proceedings. (A81-40136 18-01) Washington, DC, American Helicopter Society, 1980. 14 p. AA(U.S. Army, Aeromechanics Laboratory, Moffett Field, CA) 14 p. Refs. 9

A 1.62-m diameter rotor model was used to investigate aeromechanical stability, and the results were compared to theory. Configurations tested included: (1) a non-matched stiffness rotor as a baseline, (2) the baseline rotor with negative pitch-lag coupling, (3) the combination of negative pitch-lag coupling and structural flap-lag coupling on the baseline rotor, (4) a matched stiffness rotor, and (5) a matched stiffness rotor with negative pitch-lag coupling. The measured lead-lag regressing mode damping of the five configurations agreed well with theory, but only the matched stiffness case with negative pitch-lag coupling was able to stabilize the air resonance mode. Comparison of theory and experiment for the damping of the body modes showed significant differences that may be related to rotor inflow dynamics. (Author)

Controlled Terms: AERODYNAMIC STABILITY / *AEROELASTICITY / *AIRCRAFT STABILITY / COUPLINGS / *HELICOPTER DESIGN / HINGES / ROTARY STABILITY / *ROTARY WINGS / *ROTOR AERODYNAMICS / ROTOR SPEED / STABILITY TESTS / *VIBRATION DAMPING / VIBRATION MODE

Reference no 104

81A40154 NASA ISSUE 18 Category 5

Aeroelastic stability analysis of hingeless rotor helicopters in forward flight using blade and airframe normal modes

LYTWYN, R. T.

AHS 80-24 800000 In: American Helicopter Society, Annual Forum, 36th, Washington, DC, May 13-15, 1980, Proceedings. (A81-40136 18-01) Washington, DC, American Helicopter Society, 1980. 11 p. AA(Boeing Vertol Co., Philadelphia, PA) 11 p. Refs. 10

A mathematical computer model for evaluating aeroelastic stability of single rotor helicopters, in which blade and elastic fuselage motions are defined by means of appropriate normal modes, is described and evaluated against experimental data for aeroelastic stability in hover and in forward flight. The model utilizes a discrete representation of elastic blade modes for the definition of aeroelastic stability equations of the entire coupled rotor and fuselage system, and applies Floquet analysis for stability definition of the complete aeroelastically coupled rotor and helicopter fuselage system in forward flight. The paper describes the mathematical model structure, the analytical formulation, and a systematic damping correlation with experimental results for aeroelastic stability from model testing, full scale whirl tower studies, and Bearingless Main Rotor (BMR) flight evaluations. (Author)

Controlled Terms: *ACCELERATION (PHYSICS) / AERODYNAMIC LOADS / *AIRCRAFT MANEUVERS / AIRCRAFT STABILITY / ANGLE OF ATTACK / CONTROLLABILITY / *CRITICAL LOADING / *FIXED WINGS / FLIGHT CHARACTERISTICS / FLIGHT MECHANICS / HELICOPTER CONTROL / *KINEMATICS / *ROTARY WING AIRCRAFT

Reference no 98

81A44122 NASA ISSUE 21 Category 5
Control characteristics of a Buoyant Quad-Rotor Research Aircraft
NAGABHUSHAN, B. L.; LICHTY, D. W.; TOMLINSON, N. P.
AIAA 81-1838 810000 In: Guidance and Control Conference, Albuquerque, NM, August 19-21, 1981, Collection of Technical Papers. (A81-44076 21-12) New York, American Institute of Aeronautics and Astronautics, Inc., 1981, p. 370-377. AC(Goodyear Aerospace Corp., Defense Systems Div., Akron, OH) 8 p. Jpn. 3620

Control characteristics of a Buoyant Quad-Rotor Research Aircraft are predicted by considering such a vehicle configuration based on preliminary design. Concepts for controlling the vehicle with or without its external sling load are evaluated by using a flight dynamics simulation of the configuration. Results are presented which show the vehicle response to control inputs, wind disturbances, and power failure while hovering over a point on the ground. Typical control power and trim characteristics of the vehicle are also discussed. (Author)

Controlled Terms: AIRCRAFT CONFIGURATIONS / *AIRCRAFT CONTROL / *AIRCRAFT DESIGN / *AIRCRAFT STABILITY / FLIGHT SIMULATION / HOVERING / *ROTOR SYSTEMS RESEARCH AIRCRAFT

Reference no 99

81A41823 NASA ISSUE 19 Category 5
Helicopter theory --- Book
JOHNSON, W.
National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)
800000 Princeton, N.J, Princeton University Press, 1980. 1110 p. AA(NASA, Ames Research Center, Moffett Field, CA; U.S. Army, Washington, DC) 1110 p. Refs. 198 $95

A comprehensive presentation is made of the engineering analysis methods used in the design, development and evaluation of helicopters. After an introduction covering the fundamentals of helicopter rotors, configuration and operation, rotary wing history, and the analytical notation used in the text, the following topics are discussed: (1) vertical flight, including momentum, blade element and vortex theories, induced power, vertical drag and ground effect; (2) forward flight, including in addition to momentum and vortex theory for this mode such phenomena as rotor flapping and its higher harmonics, tip loss and root cutout, compressibility and pitch-flap coupling; (3) hover and forward flight performance assessment; (4) helicopter rotor design; (5) rotary wing aerodynamics; (6) rotary wing structural dynamics, including flutter, flap-lag dynamics ground resonance and vibration and loads; (7) helicopter aeroelasticity; (8) stability and control (flying qualities); (9) stall; and (10) noise. O.C.

Controlled Terms: *AERODYNAMIC CHARACTERISTICS / AERODYNAMIC STABILITY / AERODYNAMIC STALLING / AIRCRAFT NOISE / AIRCRAFT STABILITY / BIBLIOGRAPHIES / *FLIGHT MECHANICS / FOURIER ANALYSIS / HELICOPTER CONTROL / *HELICOPTER DESIGN / *HELICOPTER PERFORMANCE / *ROTARY WINGS / *ROTOR AERODYNAMICS / VERTICAL FLIGHT / VORTICES

Reference no 100

81A40184 NASA ISSUE 18 Category 71
Trailing edge noise from hovering rotors
KIM, Y. N.; GEORGE, A. R.
Cornell Univ., Ithaca, N. Y. (C5729333)
AHS 80-60 800000 In: American Helicopter Society, Annual Forum, 36th, Washington, DC, May 13-15, 1980, Proceedings. (A81-40136 18-01) Washington, DC, American Helicopter Society, 1980. 14 p. Army-NASA-supported research. AB(Cornell University, Ithaca, NY) 14 p. Refs. 29

A method has been developed to predict the high frequency broadband noise due to the interaction of convecting turbulent eddies with the trailing edges of a hovering rotor. The trailing edge noise from each blade was modeled as point dipole noise with spanwise loading corrections. This point dipole approximation was checked by applying the concept to a stationary airfoil in a moving medium with excellent results. In order to estimate the strength of the point dipole, the trailing edge noise theory of Amiet was used. The method was applied specifically to blade boundary layer turbulence and compared to incident atmospheric turbulence noise. The results indicate that the relative importance of these two mechanisms is related to the magnitudes of the intensity and of the length scales of the inflow and boundary layer turbulence. The results tend to fall below some available experimental data indicating that in those experiments other broadband noise sources were stronger than boundary layer-trailing edge noise. The approach which was developed is also applicable to other blade-turbulence interaction mechanisms such as local stall and tip noise. (Author)

Controlled Terms: AERODYNAMIC STALLING / AIRFOILS / *ATMOSPHERIC TURBULENCE / BLADE TIPS / COMPUTATIONAL FLUID DYNAMICS / *ENGINE NOISE / *HELICOPTERS / HIGH FREQUENCIES / *HOVERING STABILITY / *NOISE PREDICTION (AIRCRAFT) / NOISE SPECTRA / *ROTARY WINGS / *TRAILING EDGES / TURBULENCE EFFECTS / TURBULENT BOUNDARY LAYER / VORTICES

Reference no 101

81A40181 NASA ISSUE 18 Category 5
Design and development of the Model 412 helicopter
CRESAP, W. L.; MYERS, A. W.
AHS 80-56 800000 In: American Helicopter Society, Annual Forum, 36th, Washington, DC, May 13-15, 1980, Proceedings. (A81-40136 18-01) Washington, DC, American Helicopter Society, 1980. 12 p. AB(Bell Helicopter Textron, Fort Worth, TX) 12 p. Refs. 6

Using multibladed rotor technology developed with the Model 654 rotor, a four-bladed, soft-inplane rotor system has been developed for the Model 212 helicopter. The new rotor was designed to permit an increase in the allowable gross weight of the 212 and to eliminate airspeed restrictions caused by vibrations and rotor loads. The four-bladed system was chosen in lieu of a new two-bladed rotor/nodal

A flight experiment was conducted using the NASA-Army V/STOLAND UH-1H variable-stability helicopter to investigate the influence of several longitudinal-static-stability, control-augmentation, and flight-director parameters on helicopter flying qualities during terminal area operations in instrument conditions. This experiment, which was part of a joint NASA/FAA program pertaining to helicopter IFR airworthiness, was designed to corroborate and extend previous ground simulation results obtained in this program. Variations examined included stable and neutral longitudinal control position gradients, rate-damping and attitude-command augmentation, and raw data versus flight-director displays. Pilot rating results agreed excellently with the ground simulation data, indicating an adequate instrument capability with rate-damping augmentation and neutral statics and the need for pitch-roll attitude augmentation to achieve a satisfactory system

Controlled Terms: AIRCRAFT EQUIPMENT / *AIRCRAFT STABILITY / AUGMENTATION / *FLIGHT CONTROL / FLIGHT MECHANICS / FLIGHT SIMULATION / FLIGHT TESTS / *HELICOPTER CONTROL / *INSTRUMENT FLIGHT RULES / LONGITUDINAL CONTROL / *STATIC STABILITY

Reference no 95

81A46620 NASA ISSUE 22 Category 8

Full-scale wind-tunnel test of the aeroelastic stability of a bearingless main rotor

WARMBRODT, W.; MCCLOUD, J., III; SHEFFLER, M.; STALEY, J.

National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)

810000 In: American Helicopter Society, Annual Forum, 37th, New Orleans, LA, May 17-20, 1981, Proceedings. (A81-46603 22-01) Washington, DC, American Helicopter Society, 1981, p. 204-216. AB(NASA, Ames Research Center, Moffett Field, CA) AD(Boeing Vertol Co., Philadelphia, PA) 13 p. Refs. 12 Jpn. 3816

The rotor studied in the wind tunnel had previously been flight tested on a BO-105 helicopter. The investigation was conducted to determine the rotor's aeroelastic stability characteristics in hover and at airspeeds up to 143 knots. These characteristics are compared with those obtained from whirl-tower and flight tests and predictions from a digital computer simulation. It was found that the rotor was stable for all conditions tested. At constant tip speed, shaft angle, and airspeed, stability increases with blade collective pitch setting. No significant change in system damping occurred that was attributable to frequency coalescence between the rotor inplane regressing mode and the support modes. Stability levels determined in the wind tunnel were of the same magnitude and yielded the same trends as data obtained from whirl-tower and flight tests. G.R.

Controlled Terms: *AEROELASTICITY / CORRELATION / *DYNAMIC STABILITY / FLOW VELOCITY / *FULL SCALE TESTS / HOVERING STABILITY / *ROTARY WINGS / VIBRATION MODE / *WIND TUNNEL TESTS

Reference no 96

81A46619 NASA ISSUE 22 Category 8

A unified approach to the optimal design of adaptive and gain scheduled controllers to achieve minimum helicopter rotor vibration

MOLUSIS, J. A.; HAMMOND, C. E.; CLINE, J. H.

810000 In: American Helicopter Society, Annual Forum, 37th, New Orleans, LA, May 17-20, 1981, Proceedings. (A81-46603 22-01) Washington, DC, American Helicopter Society, 1981, p. 188-203. AA(Connecticut, University, Storrs, CT) AB(U.S. Army, Applied Technology Laboratory, Fort Eustis, VA) AC(U.S. Army, Structures Laboratory, Hampton, VA) 16 p. Refs. 15 Jpn. 3816

Attention is given to stochastic control theory, aspects of cautious control algorithm development, the advantages of adaptive controllers, the linear quadratic Gaussian controller, the perturbation controller, a proportional-integral controller, a combination adaptive/gain scheduled controller, aspects of controller testing, the application of adaptive control algorithms, the application of gain scheduled controllers, and vibration reduction results. Six different control laws are developed. Four of them are implemented and evaluated in the Langley Transonic Dynamics wind tunnel. The obtained results provide the means for using control design to achieve theoretically minimum rotor vibration throughout the flight envelope. C.R.

Controlled Terms: *ACTIVE CONTROL / *ADAPTIVE CONTROL / *AIRCRAFT CONTROL / ALGORITHMS / *CONTROL THEORY / *HELICOPTER DESIGN / PERTURBATION THEORY / *ROTARY WINGS / STOCHASTIC PROCESSES / VIBRATION DAMPING

Reference no 97

81A44554 NASA ISSUE 21 Category 8

Kinematic properties of rotary-wing and fixed-wing aircraft in steady coordinated high-g turns

CHEN, R. T. N.

National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)

AIAA PAPER 81-1855 810800 American Institute of Aeronautics and Astronautics, Atmospheric Flight Mechanics Conference, Albuquerque, NM, Aug. 19-21, 1981, 15 p. AA(NASA, Ames Research Center, Moffett Field, CA) 15 p. Refs. 24 Jpn. 3625

An analytical approach to the study of flight dynamics of aircraft operating in a high-angle-of-attack flight regime and of helicopters operating in extreme thrust conditions is presented. Steady coordinated high-g turns are used to establish the initial equilibrium flight conditions near stall angles of attack. The kinematic properties of the aircraft in steady coordinated turns are examined: in high-g turns, pitch rate (independent of the angle of attack) is of a much larger magnitude than roll and yaw rate; a substantial roll rate is found to develop in steep turns for all angles of attack; the angle of attack also has a significant effect on the pitch attitude, with decreasing influence as the normal load factor increases. The exact small disturbance equations of motion of the aircraft in general steady turns are also developed for application to both rotary-wing and fixed-wing aircraft in extreme conditions. These equations are in a first-order, vector-matrix format, and are thus compatible with many efficient software packages developed in modern system theory. J.F.

A steep coordinated helical turn at extreme angles of attack with inherent sideslip is of primary interest in this study. Unlike fixed-wing aircraft, the helicopter in a steady coordinated turn will inherently sideslip. A set of exact kinematic equations describing this motion in steady helical turns has been developed, and a rational definition for the load factor that best characterizes a coordinated turn for a helicopter has been proposed. An analysis has also been completed on the effects of sideslip on the kinematic relationships in a coordinated turn which is based on new closed-form solutions which relate the aircraft angular rates and pitch and roll attitudes to the turn parameters, angle of attack, and sideslip. The results show that the bank angle of the aircraft can differ markedly from the tilt angle of the normal load factor and that the normal load factor can also differ substantially from the accelerometer reading along the vertical body axis of the aircraft. Generally, sideslip has a strong influence on the pitch attitude and roll rate of the helicopter. The study also indicates that pitch rate is independent of angle of attack in a coordinated turn and that in the absence of sideslip, angular rates about the stability axes are independent of the aerodynamic characteristics of the aircraft. (Author)

Controlled Terms: AERODYNAMIC CHARACTERISTICS / AERODYNAMIC LOADS / *AIRCRAFT MANEUVERS / *ANGLE OF ATTACK / HELICOPTER CONTROL / *HELICOPTER PERFORMANCE / *KINEMATICS / PITCH (INCLINATION) / ROLLING MOMENTS / *SIDESLIP / TURNING FLIGHT

Reference no 92

81A46627 NASA ISSUE 22 Category 8

Handling qualities of the SH-60B Seahawk

ALANSKY, I.; FAULL, R.; SKONIECZNY, J.

810000 In: American Helicopter Society, Annual Forum, 37th, New Orleans, LA, May 17-20, 1981, Proceedings. (A81-46603 22-01) Washington, DC, American Helicopter Society, 1981, p. 288-306. AC(United Technologies Corp., Sikorsky Aircraft Div., Stratford, CT) 19 p. Jpn. 3817

A review is provided of the flying qualities, the automatic flight control system, and the developmental testing of the U.S. Navy's most advanced helicopter. Primary mission of the SH-60B Seahawk are related to anti-submarine warfare and anti-ship surveillance and targeting. Its secondary missions include search and rescue, communications relay, medical evaluation, and vertical replenishment. The operational considerations for the SH-60B helicopter require that these missions be performed from nonaviation type ships in conditions reaching upper sea-state 5. An examination is conducted of the primary differences between the Seahawk and the UH-60A Black Hawk. The differences result primarily from the Light Airborne Multi-Purpose Systems (LAMPS) MK III mission requirements. Handling qualities design considerations are discussed, taking into account the aerodynamic effects of mission equipment, a SH-60B simulation math model, control requirements, static stability characteristics, shipboard landings, and hydrodynamic stability considerations. G.R.

Controlled Terms: AERODYNAMIC CHARACTERISTICS / AIRCRAFT CONFIGURATIONS / AIRCRAFT LANDING / AUTOMATIC FLIGHT CONTROL / *CONTROLLABILITY / *FLIGHT CHARACTERISTICS / *HELICOPTER DESIGN / *MILITARY HELICOPTERS / NAVY / TIME RESPONSE

Reference no 93

81A46626 NASA ISSUE 22 Category 8

A preliminary flight investigation of cross-coupling and lateral damping for nap-of-the-earth helicopter operations

CORLISS, L. D.; CARICO, G. D.

810000 In: American Helicopter Society, Annual Forum, 37th, New Orleans, LA, May 17-20, 1981, Proceedings. (A81-46603 22-01) Washington, DC, American Helicopter Society, 1981, p. 276-287. AB(U.S. Army, Aeromechanics Laboratory, Moffett Field, CA) 12 p. Refs. 9 Jpn. 3816

A helicopter in-flight simulation was conducted to look at the effects of variations in roll damping, roll sensitivity, and pitch and roll rate cross-coupling on helicopter flying qualities for NOE operations. The experiment utilized the Ames UH-1H helicopter in-flight simulator, which is equipped with the V/STOLAND avionics system. The response envelope of this vehicle allowed for the simulation of configurations in the low-to-moderate damping and sensitivity range. A visual, low-level slalom course was set up, using the 1000-ft markers of an 8000-ft runway to evaluate the various configurations. Test results are shown in terms of Cooper-Harper pilot ratings. These results show good consistency with previous ground simulator results and with some elements of flying qualities criteria, such as those of MIL-F-83300 and MIL-H-8501A. (Author)

Controlled Terms: AERODYNAMIC STABILITY / COMPUTER TECHNIQUES / *CROSS COUPLING / *DAMPING / DATA ACQUISITION / DIGITAL TECHNIQUES / FLIGHT CHARACTERISTICS / *FLIGHT SIMULATION / *HELICOPTER PERFORMANCE / *LATERAL STABILITY / *NAP-OF-THE-EARTH NAVIGATION

Reference no 94

81A46623 NASA ISSUE 22 Category 8

A flight investigation of static stability, control augmentation, and flight director influences on helicopter IFR handling qualities

LEBACQZ, J. V.; WEBER, J. M.; CORLISS, L. D.

National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)

810000 In: American Helicopter Society, Annual Forum, 37th, New Orleans, LA, May 17-20, 1981, Proceedings. (A81-46603 22-01) Washington, DC, American Helicopter Society, 1981, p. 237-251. AB(NASA, Ames Research Center, Moffett Field, CA) AC(U.S. Army, Aeromechanics Laboratory, Moffett Field, CA) 15 p. Refs. 15 Jpn. 3816

Reference no 87
82N15013# NASA ISSUE 6 Category 2
Flag-lag-torsional dynamics or extensional and inextensional rotor blades in hover and in forward flight
Semiannual Progress Report, Jul. - Dec 1981
CRESPODASILVA, M. R. M.
Cincinnati Univ., Ohio. (CP730085)
NASA-CR-165078 811200 NAG2-38 Dept. of Aerospace Engineering and Applied Mechanics. 6 p. Jpn. 717 HC A02/MF A01

The differential equations describing the flap-lag-torsional motion of a flexible rotor blade including third-order nonlinearities were derived for hover and forward flight. Making use of the two boundary conditions, those equations were reduced to a set of three integro partial differential equations written in terms of the flexural deflections and the torsional variable. L.F.M.

Controlled Terms: AERODYNAMICS / ANALYSIS (MATHEMATICS) / DYNAMIC STABILITY / *EQUATIONS OF MOTION / *HELICOPTERS / *HOVERING STABILITY / *ROTARY WINGS / ROTOR BLADES (TURBOMACHINERY) / *TORSIONAL STRESS

Reference no 88
82N13137# NASA ISSUE 4 Category 5
System identification helicopter parameters. Determination from flight tests, phase 2
systemidentifizierung drehfluegler^kennwertermittlung aus flugmessungen (phase 2)
KLOSTER, M.; ATTLFELLNER, S.
Messerschmitt-Boelkow-Blohm G.m.b.H., Munich (West Germany). (MT618891)
BMVG-FBWT-80-12 800000 Unternehmensbereich Drehfluegler und Verkehr. Sponsored by Bundesministerium der Verteidigung In GERMAN; ENGLISH summary 92 p. Bonn Bundesministerium der Verteidigung Jpn. 446 HC A05/MF A01; DOKZENTw, Bonn DM 30

A parameter identification program for a hingeless rotor helicopter is considered. Flight conditions were selected with increasing instability, i.e., hover and level flight at maximum speed; with maximum weight and with a rearward center of gravity. A strap down system was chosen to provide the attitude feedback control necessary for proper identification. The control input signals were optimized for the unstabilized helicopter. Calculations in the time and frequency domains show that special distributions in the power spectrum of the input signals are needed for optimizing the closed loop system. The identified derivatives and the smoothened time histories from flight tests are compared with the identification results of linear and nonlinear simulations and of the quasistatic theory. Author (ESA)

Controlled Terms: *AERODYNAMIC STABILITY / ATTITUDE INDICATORS / COMPUTERIZED SIMULATION / FEEDBACK CONTROL / *HELICOPTER PERFORMANCE / *PARAMETER IDENTIFICATION / *RIGID ROTORS / ROTARY WINGS

Reference no 89
82A12045 NASA ISSUE 2 Category 5
Numerical treatment of helicopter rotor stability problems
VEPA, R.; BALASUBRAMANIAN, T. S.
800000 In: Emerging technologies in aerospace structures, design, structural dynamics and materials; Proceedings of the Aerospace Conference, San Francisco, CA, August 13-15, 1980. (A82-12026 02-39) New York, American Society of Mechanical Engineers, 1980, p. 307-319. Research supported by the Ministry of Defence. AB(National Aeronautical Laboratory, Bangalore, India) 13 p. Refs. 14

The problem of calculating the Floquet transition matrix for parametric stability problems is considered. A new method of calculating the transition matrix with a minimum number of time steps is described. The method is shown to be extremely efficient for a wide class of helicopter stability problems, including flapping stability of a helicopter rotor, helicopter ground resonance with a nonisotropic rotor and pitch-flap-bending stability of a helicopter rotor in forward flight. The relationship of the method to the classical method of averaging is pointed out. (Author)

Controlled Terms: *AERODYNAMIC STABILITY / *AIRCRAFT STABILITY / DYNAMIC RESPONSE / EQUATIONS OF MOTION / *FLOQUET THEOREM / *HELICOPTER PERFORMANCE / MATRICES (MATHEMATICS) / *NUMERICAL ANALYSIS / *ROTARY WINGS / *ROTOR AERODYNAMICS / RUNGE-KUTTA METHOD / TRANSFORMATIONS (MATHEMATICS)

Reference no 90
82N12042# NASA ISSUE 3 Category 2
Experimental and analytical studies of a model helicopter rotor in hover
CARADONNA, F. X.; TUNG, C.
National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)
NASA-TM-81232; A-8332; USAAVRADCOM-TR-81-A-23 810900 Prepared jointly with Army Aviation Research and Development Command Presented at the 6th European Rotorcraft and Powered Lift Aircraft Forum, Bristol, England, 16-19 Sep. 1980 61 p. Jpn. 289 HC A04/MF A01

A benchmark test to aid the development of various rotor performance codes was conducted. Simultaneous blade pressure measurements and tip vortex surveys were made for a wide range of tip Mach numbers including the transonic flow regime. The measured tip vortex strength and geometry permit effective blade loading predictions when used as input to a prescribed wake lifting surface code. It is also shown that with proper inflow and boundary layer modeling, the supercritical flow regime can be accurately predicted. J.M.S.

Controlled Terms: *BLADE TIPS / FLOW GEOMETRY / *HELICOPTER WAKES / *HOVERING / PREDICTION ANALYSIS TECHNIQUES / PRESSURE DISTRIBUTION / *ROTOR AERODYNAMICS / TRANSONIC FLOW / *VORTICES

Reference no 91
81A46644 NASA ISSUE 22 Category 8
Influence of sideslip on the kinematics of the helicopter in steady coordinated turns
CHEN, R. T. N.; JESKE, J. A.
National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)
810000 In: American Helicopter Society, Annual Forum, 37th, New Orleans, LA, May 17-20, 1981, Proceedings. (A81-46603 22-01) Washington, DC, American Helicopter Society, 1981, p. 463-477. AB(NASA, Ames Research Center, Moffett Field, CA) 15 p. Refs. 16 Jpn. 3817

A computer-flight-testing (CFT) program for helicopters to evaluate helicopter dynamics and handling and control qualities is described. The nonlinear six degrees of freedom helicopter model is driven by control inputs generated by a specially developed control model (or 'pseudo pilot') to eliminate problems in estimating control inputs during maneuvering flight. This is an adaptation of a linear optimal control model as used in human factor analysis. The helicopter model is based on two-dimensional strip aerodynamics steady-state rotor blade dynamics using only out-of-plane bending mode shapes, which are suitable for various types of rotor articulation. The 'pilot' model consists of a flight path generation (FPG)-model and a stabilitization (STAB)-model. The FPG-model is based on linearized system dynamics using terminal optimal control, generating both required flight path and the control inputs to achieve it. These controls are input into the helicopter model. The two flight paths are compared, and differences are fed back to the STAB-model to generate corrective control inputs of such a nature that the helicopter-model-generated flight path tracks the required flight path generated by the FPG-model. Also the STAB-model is based on linearized system dynamics. As an example, two flare maneuvers are 'flown', and the results discussed. The pseudo-pilot model performs well, provided that helicopter dynamics do not change much during a specific maneuver. J.M.S.

Controlled Terms: *AERODYNAMIC STABILITY / *AUTOMATIC FLIGHT CONTROL / *CONTROL STABILITY / FLIGHT PATHS / *HELICOPTER CONTROL / LINEAR SYSTEMS / *MANEUVERABILITY / OPTIMAL CONTROL / TERMINAL GUIDANCE

Reference no 84

82N18121# NASA ISSUE 9 Category 2
A simplified approach to the free wake analysis of a hovering rotor
MILLER, R. H.
Massachusetts Inst. of Tech., Cambridge. (MJ700802)
810000 In DGLR Seventh European Rotorcraft and Powered Lift Aircraft Forum 15 p (SEE N82-18119 09-01) 15 p. Jpn. 1167 HC A99/MF A01

Rotor wake geometry and the number of blades are analyzed. Simplified approaches to the free wake aerodynamic analysis of hovering rotors permitting rapid evaluation of rotor aerodynamic characteristics are discussed. Analytical results are compared with existing measurements of blade bound circulation and wake geometry. N.W.

Controlled Terms: AERODYNAMIC CHARACTERISTICS / HELICOPTER PERFORMANCE / *HELICOPTER WAKES / *HOVERING / *ROTARY WINGS / *ROTOR AERODYNAMICS / *ROTOR BLADES (TURBOMACHINERY) / ROTORS

Reference no 85

82A17861 NASA ISSUE 6 Category 8
Dynamic stability of a buoyant quad-rotor aircraft --- for airlifting payloads externally on a sling
NAGABHUSHAN, B. L.; TOMLINSON, N. P.
Goodyear Aerospace Corp., Akron, Ohio. (G2570146)
AIAA PAPER 82-0242 820100 NAS2-10777 American Institute of Aeronautics and Astronautics, Aerospace Sciences Meeting, 20th, Orlando, FL, Jan. 11-14, 1982, 10 p. AB(Goodyear Aerospace Corp., Defense Systems Div., Akron, OH) 10 p. Refs. 13

Stability characteristics of a buoyant quad-rotor aircraft (BQRA) in hover and forward flight are examined by considering linear, state-variable, and nonlinear flight simulation models of such a configuration. The effects of carrying a sling load on the vehicle dynamics is predicted by considering a coupled model of the two bodies. Inherent stability characteristics of the vehicle are analyzed and compared with those of a helicopter and an airship in free flight. Typical operational conditions that could lead to vehicle instability are described in the flight envelope of interest. (Author)

Controlled Terms: AERODYNAMIC LOADS / *AERODYNAMIC STABILITY / AIRCRAFT CONFIGURATIONS / CIVIL AVIATION / *FLIGHT SIMULATION / *HOVERING STABILITY / LINEAR SYSTEMS / MATHEMATICAL MODELS / MILITARY AIRCRAFT / NONLINEAR SYSTEMS / PERFORMANCE PREDICTION / *ROTORCRAFT AIRCRAFT / VERTICAL LANDING / VERTICAL TAKEOFF

Reference no 86

82N17158# NASA ISSUE 8 Category 5
The armed helicopter in air to air missions
WEILAND, E.
Messerschmitt-Boelkow-Blohm G.m.b.H., Ottobrunn (West Germany). (MT620643)
MBB-UD-317-81-O 810000 Betriebsbereich. Presented at Royal Aeronautical Soc. Rotorcraft Section Symp., London, 4 Feb. 1981 18 p. Jpn. 1028 HC A02/MF A01

The antihelicopter-helicopter must be superior in performance and agility. The basic vehicle must be supplemented by a specialized armament demonstrating a combination of guns and fire and forget air to air missiles. In order to ensure a system superior to the possible enemy gunship, the helicopter needs to be designed to take full advantage of the inherent performance and agility potential of the basic helicopter elements. The agility aspect of the rotorcraft inherent characteristics of the major rotorcraft components are discussed and a possibility for a practical solution is indicated. Author

Controlled Terms: *AERODYNAMIC CONFIGURATIONS / AIRCRAFT SURVIVABILITY / ATTACK AIRCRAFT / *HELICOPTER DESIGN / LATERAL CONTROL / *MANEUVERABILITY / *MILITARY HELICOPTERS / MOMENTS OF INERTIA / ROTOR AERODYNAMICS / WEAPON SYSTEMS

complexity of the problem both in hovering and forward flight are discussed. A simplified approach to a free wake analysis of rotors in hover is discussed and the analytical results compared with experimental data for two and four bladed rotors. (Author)

Controlled Terms: ACOUSTIC PROPERTIES / CIRCULATION / COMPUTATIONAL FLUID DYNAMICS / *HELICOPTER WAKES / *HOVERING / OPTIMIZATION / *PERFORMANCE PREDICTION / *ROTARY WINGS / *ROTOR AERODYNAMICS / *VORTICES / WING LOADING

Reference no 80

82N20178# NASA ISSUE 11 Category 5

A surface singularity method for rotors in hover or climb
Final Report, 29 Mar. 1979 - 30 Sep. 1981
SUMMA, J.; MASKEW, B.
Analytical Methods, Inc., Redmond, Wash. (AV021240)
AD-A109687; AMI-8103; USAAVRADCOM-TR-81-D-23 811200 DAAJ02-76-C-0069 ; DA PROJ. 1F2-62209-AH-76 142 p. Jpn. 1464 HC A07/MF A01

A surface singularity potential flow code, ROTAIR, has been assembled for the calculation of detailed surface pressures on rotors in hover or climb. This method is basically a marriage of two codes: the fixed-wing surface singularity doublet code and the rotor lifting-surface code. The program includes a tip vortex separation model. Also, the rotor tip surface is panelled so that pressures are calculated right around the tip-edge surface. Preliminary calculations have verified the capabilities of the program for computing blade surface properties in the presence of a close-vortex passage. Additionally, calculated pressure distributions compare favorably with experimental data for a low aspect ratio two-bladed rotor, and the calculated circulation distribution is consistent with that computed earlier with the lifting-surface code. Finally, the newly developed far-wake doublet model promises to keep computing efforts practical. Author (GRA)

Controlled Terms: *AERODYNAMIC LOADS / AERODYNAMIC STABILITY / *CLIMBING FLIGHT / COMPUTER PROGRAMS / FLUTTER / FORCE DISTRIBUTION / *HOVERING / MATHEMATICAL MODELS / *ROTARY WINGS / *ROTOR AERODYNAMICS / *WING LOADING

Reference no 81

82N18159# NASA ISSUE 9 Category 5

Hover tests of a model H-force rotor
VELKOFF, H. R.
Ohio State Univ., Columbus. (OM593208)
810000 NCA2-OR565-001 Dept. of Mechanical Engineering. In DGLR Seventh European Rotorcraft and Powered Lift Aircraft Forum 19 p (SEE N82-18119 09-01) 19 p. Jpn. 1172 HC A99/MF A01

The potential of using tip vanes at the ends of helicopter rotor blades to obtain a controllable H-force is considered. The addition of vanes placed perpendicular to the blade tips can be used to obtain an inplane force. By varying the angle of the vanes, a radial force can be created which can be controllable in azimuth position. Such a force could be used to provide translational motion of the rotor and aircraft without the requirement for rotor tilting. In addition, an H-force generated at high flight speed could be used as a propulsive force in a matter similar to a propeller. The force generated by the vanes could also affect the aircraft's stability characteristics. The H-force could also modify rotor performance in hovering since they could be thought to act as a virtual shroud. Tests were run with a model rotor which has a 6 foot diameter with a 3 inch chord blade. Test data are presented on the effects of various tip-vane configurations on the hovering figure of merit. The extreme sensitivity of the performance to vane arrangement is shown. Author

Controlled Terms: ATTITUDE STABILITY / CONTROL SURFACES / FIGURE OF MERIT / FINS / GAUGE INVARIANCE / HELICOPTER PERFORMANCE / HOVERING / *HOVERING STABILITY / *ROTARY WINGS / *ROTOR AERODYNAMICS / *VANES / *WING TIPS

Reference no 82

82N18155# NASA ISSUE 9 Category 5

Preliminary investigation into the addition of auxiliary longitudinal thrust on helicopter agility
LEGGE, P. J.; FORESTCUE, P. W.; TAYLOR, P.
Royal Aircraft Establishment, Farnborough (England). (R2785060)
810000 In DGLR Seventh European Rotorcraft and Powered Lift Aircraft Forum 17 p (SEE N82-18119 09-01) AB(Southampton Univ.) AC(Southampton Univ.) 17 p. Jpn. 1172 HC A99/MF A01

The agility requirements in specific maneuvers performed by military helicopters are investigated. A simple mathematical simulation of the helicopter accelerating and decelerating longitudinally is used to examine the advantages of auxiliary thrust. For helicopters operating in the Nap-of-Earth environment the longitudinal acceleration and deceleration performance is shown to be unique' important. Large improvements in agility, measured by a performance function, are obtained by adding auxiliary thrust. The need for the helicopter to change its attitude is drastically reduced, and the pilot workload improved. The type of flight profile used is also examined, using a nondimensional Froude number. There was no benefit in a 'maximum effort' flight profile, which consequently further improves the pilot workload. Auxiliary thrust improved agility performance by increasing the helicopter's ability to change position while maintaining precise attitude control. J.M.S.

Controlled Terms: ATTITUDE CONTROL / AUXILIARY PROPULSION / *LONGITUDINAL CONTROL / *MANEUVERABILITY / *MILITARY HELICOPTERS / PERFORMANCE PREDICTION

Reference no 83

82N18153# NASA ISSUE 9 Category 8

A control model for maneuvering flight for application to a computer-flight testing program
HAVERIDINGS, H.
National Aerospace Lab., Amsterdam (Netherlands). (NE736790)
810000 In DGLR Seventh European Rotorcraft and Powered Lift Aircraft Forum 16 p (SEE N82-18119 09-01) 16 p. Jpn. 1171 HC A99/MF A01

A computational procedure employing a stochastic learning method in conjunction with dynamic simulation of helicopter flight and weapon system operation was used to derive helicopter maneuvering strategies. The derived strategies maximize either survival or kill probability and are in the form of a feedback control based upon threat visual or warning system cues. Maneuverability parameters implicit in the strategy development include maximum longitudinal acceleration and deceleration, maximum sustained and transient load factor turn rate at forward speed, and maximum pedal turn rate and lateral acceleration at hover. Results are presented in terms of probability of skill for all combat initial conditions for two threat categories. T.M.

Controlled Terms: AIRCRAFT RELIABILITY / *COMBAT / CONTROLLABILITY / *EFFECTIVENESS / FEEDBACK CONTROL / HELICOPTER CONTROL / *HELICOPTER PERFORMANCE / *MANEUVERABILITY / *MILITARY HELICOPTERS / WEAPON SYSTEMS

Reference no 76
82N23209# NASA ISSUE 14 Category 3
VTOL and VSTOL handling qualities specifications, an overview of the current status
GOLDSTEIN, K. W.
Naval Air Development Center, Warminster, Pa. (NO000154)
820400 In NASA, Ames Research Center Helicopter Handling Qualities p 1-7 (SEE N82-23208 14-03) 7 p. Jpn. 1898 HC A11/MF A01

The highlights of a comparative analysis between the current helicopter and VSTOL specifications and four representative rotary wing aircraft are presented. Longitudinal, lateral, and directional control power and dynamic stability characteristics were analyzed for hovering conditions. Forward flight static and dynamic stability were analyzed for the longitudinal and lateral-directional axes. Results of the analyses in terms of the applicability/utility of the MIL-H-8501A criteria are presented for each of the above areas. The review of the MIL-H-8301A criteria against those in MIL-F-83300 and AGARD 577 indicate many areas in which MIL-H-8501A does not give adequate design guidance. T.M.

Controlled Terms: *AIRCRAFT SPECIFICATIONS / ANGLE OF ATTACK / ATTITUDE CONTROL / CONTROL STABILITY / *CONTROLLABILITY / FLIGHT CONTROL / *HELICOPTER CONTROL / HOVERING / LATERAL CONTROL / LONGITUDINAL CONTROL / *MILITARY HELICOPTERS / *V/STOL AIRCRAFT

Reference no 77
82N23208# NASA ISSUE 14 Category 3
Helicopter Handling Qualities
National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)
NASA-CP-2219; A-8891; NAS 1.55:2219 820400 Proceedings of the special meeting held at Moffett Field, Calif., 14-15 Apr. 1982; sponsored by the American Helicopter Society 243 p. Jpn. 1898 HC A11/MF A01

Helicopters are used by the military and civilian communities for a variety of tasks and must be capable of operating in poor weather conditions and at night. Accompanying extended helicopter operations is a significant increase in pilot workload and a need for better handling qualities. An overview of the status and problems in the development and specification of helicopter handling-qualities criteria is presented. Topics for future research efforts by government and industry are highlighted. For individual titles, see N82-23209 through N82-23230

Controlled Terms: AIRCRAFT INSTRUMENTS / AIRCRAFT MANEUVERS / AIRCRAFT RELIABILITY / *AIRCRAFT SPECIFICATIONS / AIRCRAFT SURVIVABILITY / ALL-WEATHER AIR NAVIGATION / AUTOMATIC FLIGHT CONTROL / *AVIONICS / *COCKPITS / COMBAT / *CONFERENCES / CONTROL BOARDS / *CONTROLLABILITY / DISPLAY DEVICES / FLIGHT CONTROL / *HELICOPTER CONTROL / HELICOPTER PERFORMANCE / *MANEUVERABILITY / *NAP-OF-THE-EARTH NAVIGATION / *NIGHT FLIGHTS (AIRCRAFT) / RADAR NAVIGATION / STABILITY AUGMENTATION

Reference no 78
82N22251# NASA ISSUE 13 Category 5
Performance testing of a main rotor system for a utility helicopter at 1/4 scale
BERRY, J. D.
National Aeronautics and Space Administration. Langley Research Center, Hampton, Va. (ND210491)
NASA-TM-83274; L-15015; NAS 1.15:83274; AVRADCOM-TR-82-B-3 820400 DA PROJ. 1L1-61102-AH-45 AA(Army Aviation Research and Development Command, Langley, Va.) 49 p. Jpn. 1758 HC A03/MF A01

Two rotor systems for the UH-1 helicopter were tested in hover and forward flight. The baseline system was a dynamically scaled model of the current rotor system, while the other system was designed for advanced performance. In hover out of ground effect, the advanced rotor system shows improvements up to 10 percent in the figure of merit and improvements in thrust up to 7 percent. In forward flight, the advanced rotor system demonstrated reductions in required torque throughout the range of conditions tested, with reductions up to 17 percent occurring at the higher advance ratios and higher lift values tested. T.M.

Controlled Terms: AERODYNAMIC CHARACTERISTICS / *GROUND EFFECT (AERODYNAMICS) / HORIZONTAL FLIGHT / *HOVERING / *ROTARY WINGS / *ROTOR AERODYNAMICS / SCALE MODELS / *UH-1 HELICOPTER / WIND TUNNEL MODELS / WIND TUNNEL TESTS

Reference no 79
82A22044 NASA ISSUE 8 Category 2
Rotor hovering performance using the method of fast free wake analysis
MILLER, R. H.
AIAA PAPER 82-0094 820100 American Institute of Aeronautics and Astronautics, Aerospace Sciences Meeting, 20th, Orlando, FL, Jan. 11-14, 1982, 7 p. AA(MIT, Cambridge, MA) 7 p. Refs. 11

Experimental investigations have shown that the bound circulation distribution on a rotor blade is critically dependent on the location of the vortices in the wake. Consequently in order to optimize rotor performance, compute blade loads, and determine acoustic signatures, it is necessary to use analytical methods which are capable of predicting wake geometry. The factors which contribute to the

Reference no 128

81N21074# NASA ISSUE 12 Category 5
Study of oscillatory and flight dynamic behavior of helicopters in atmospheric turbulence
DAHL, H.; WEGER, D.
Messerschmidt-Boelkow G.m.b.H., Munich (West Germany). (MT623300)
BMVG-FBWT-80-4 800000 Sponsored by Bundesminister der Verteidigung
In GERMAN; ENGLISH summary 72 p. Bonn Bundesminister der Verteidigung
Jpn. 1583 HC A04/MF A01

A statistical model was developed which takes the nonuniform nature of the gust velocity distribution into account together with the dynamic transient oscillation of the rotor blades. Because gust effects depend strongly on the helicopter's control behavior, an analytic pilot function is included in the considerations. Hingeless and hinged rotor systems are compared. Physiological factors and data acquisition are also discussed. The results enable gust reduction systems to be incorporated in the design of helicopters. Author (ESA)

Controlled Terms: AIRCRAFT MODELS / *ATMOSPHERIC TURBULENCE / DATA ACQUISITION / FLAPPING HINGES / FLIGHT CHARACTERISTICS / *FLIGHT SIMULATION / GUST LOADS / *HELICOPTER CONTROL / PHYSIOLOGICAL FACTORS / PILOT PERFORMANCE / ROTOR AERODYNAMICS / TRANSIENT OSCILLATIONS

Reference no 129

81N20745# NASA ISSUE 11 Category 61
The Data from Aeromechanics Test and Analytics, Management and Analysis Package (DATAMAP). Volume 1: User's manual
Final Technical Report
PHILBRICK, R. B.
Textron Bell Helicopter, Fort Worth, Tex. (TV739419)
AD-A095188; BHT-699-099-025-VOL-1; USAAVRADCOM-TR-80-D-30-A 801200
DAAK51-79-C-0015; DA PROJ. 1L1-62209-AH-76 280 p. Jpn. 1534 HC A13/MF A01

The Data from Aeromechanics Test and Analytics Management and Analysis Package (DATAMAP) was designed and programmed as a computer software tool for data management and processing of large, time-based data bases. Particular attention is given to rotorcraft-related analyses. The package will process data stored in two basic formats. The first format is that used for the Operational Loads Survey (OLS) test data based and anticipated for use in planned flight test programs. The second format is more general; it accommodates various data structures common to analytical data bases. This particular input capability is demonstrated by an interface between the Rotorcraft Flight Simulation Program (C81) and DATAMAP. The package transfers selected data to a large, direct access disc file and maintains the data on a semi-permanent basis. Data are retrieved from this file, processed, and displayed interactively or in batch. Plot output is generated on a Tektronix 4014 or an incremental plotter (e.g., Calcomp). A small sample of available processing options includes amplitude spectrum, harmonic analysis, digital filtering, auto-spectral density, frequency response function, acoustic analyses, and blade static pressure and normal force coefficient derivations. This program will accommodate data from multiple sensors simultaneously for processing of functions with two geometric independent variables (e.g., chord and radius). The package is written entirely in FORTRAN. Package specifications require nonstandard FORTRAN coding to be used, but the package has been made as easily transportable as possible. GRA

Controlled Terms: *COMPUTER GRAPHICS / *DATA PROCESSING / DIGITAL SYSTEMS / HARMONIC ANALYSIS / *INFORMATION MANAGEMENT / SIGNAL PROCESSING / USER MANUALS (COMPUTER PROGRAMS)

Reference no 130

81A20537 NASA ISSUE 7 Category 2
Prediction of tilt rotor outwash
WERNICKE, R. K.
AIAA PAPER 81-0013 810100 American Institute of Aeronautics and Astronautics, Aerospace Sciences Meeting, 19th, St. Louis, Mo., Jan. 12-15, 1981, 12 p. AA(Bell Helicopter Textron, Fort Worth, Tex.) 12 p. Refs. 6 Jpn. 988

Increased disc loadings (the ratio of gross weight to rotor area) have made working beneath a hovering aircraft difficult. A review of downwash/outwash data for various rotor craft with disc loadings of 4-40 psf indicates that the flow field in the area under the craft at 1.2-1.6 rotor radii from the center of the rotor produces the highest overturning moments. If it is unsteady, however, the downwash may become upsetting at a relatively low velocity, suggesting that parameters other than disc loading also effect downwash pressure and, consequently, overturning moments. Although the flow fields beneath the rotors of tilt rotor aircraft like the XV-15 are essentially identical to those of a single-rotor craft, they create an additional outwash from the vehicle's nose and tail. This outwash is problematic though at disc loadings greater than 25 psf. R.S.

Controlled Terms: AERODYNAMIC LOADS / DOWNWASH / *HOVERING STABILITY / PERFORMANCE PREDICTION / *ROTOR AERODYNAMICS / *TILT ROTOR AIRCRAFT / VELOCITY DISTRIBUTION / VERTICAL TAKEOFF AIRCRAFT

Reference no 131

81A18600 NASA ISSUE 6 Category 2
Practical aerodynamics of the helicopter Mi-6A --- Russian book
Prakticheskaia aerodinamika vertoleta Mi-6A
LALETIN, K. N.; ARTAMONOV, L. T.
800000 Moscow, Izdatel'stvo Transport, 1980. 168 p. In Russian. 168 p. Refs. 14

The book deals with some characteristic features of the rotor and airframe of the Mi-6A helicopter. The balancing, stability, and controllability of the helicopter in unsteady and some special modes of flight are examined. Attention is given to the behavior of the helicopter in flights with a suspended load and in emergency situations, such as failure of one of the two engines, failure of the tail rotor, etc. V.P.

Controlled Terms: *AERODYNAMIC STABILITY / AIRCRAFT LANDING / AIRCRAFT MANEUVERS / AUTOMATIC PILOTS / BALANCING / CONTROLLABILITY / EMERGENCIES / *FLIGHT CHARACTERISTICS / GROUND EFFECT (AERODYNAMICS) / *HELICOPTER PERFORMANCE / HOVERING / *ROTARY WINGS / *ROTOR AERODYNAMICS / VERTICAL MOTION

Reference no 132

81N18052# NASA ISSUE 9 Category 5

Preliminary airworthiness evaluation OH-58C helicopter configured with a mast mounted sight

Final Report

BISHOP, J. A.; HANKS, M. L.; BENSON, T. P.; ARRIGO, A. M.

Army Aviation Engineering Flight Activity, Edwards AFB, Calif. (AZ102498)

AD-A094172; USAAEFA-78-09 800500 76 p. Jpn. 1148 HC A05/MF A01

The United States Army Aviation Engineering Flight Activity conducted a preliminary airworthiness evaluation of the OH-58C helicopter with a mast mounted sight and Bell Helicopter, Textron Model 570B three-axis stability and control augmentation system. The evaluation was completed in two phases at the Bell Helicopter Engineering Flight Research Center, Arlington, Texas (elevation 630 ft). Phase 1 consisted of an evaluation of a dummy mast mounted sight installed on an instrumented helicopter. This phase was conducted between 15 and 30 October 1980 and 12 flights were flown for a total of 9.7 productive hours. Phase 2 consisted of a qualitative assessment of the handling qualities of the OH-58C configured with the operational Rockwell International sight. This phase was completed on 30 November 1980 in two flights for 1.5 productive test flight hours. The overall evaluation indicated that the OH-58C handling qualities with installed mast mounted sight and three-axis SCAS were satisfactory within the flight envelope tested. No problems were noted that will prevent future operational testing of the system. The addition of a three-axis SCAS significantly improved the OH-58C handling qualities, particularly in low speed flight, and is an enhancing characteristic. GRA

Controlled Terms: *AERODYNAMIC STABILITY / *AIRCRAFT CONTROL / *AIRCRAFT RELIABILITY / FLIGHT OPERATIONS / *FLIGHT TESTS / *HELICOPTERS / MECHANICAL DRIVES / VIBRATION

Reference no 133

81N18028# NASA ISSUE 9 Category 2

Mult.cyclic controllable twist rotor data analysis

Final Report

WEI, F. S.; WEISBRICH, A. L.

Kaman Aerospace Corp., Bloomfield, Conn. (KC616275)

NASA-CR-152251; R-1562 790115 NAS2-8726 170 p. Jpn. 1144 HC A08/MF A01

Rsults provide functional relationship between rotor performance, blade vibratory loads and dual control settings and indicate that multicyclic control produced significant reductions in blade flatwise bending moments and blade root actuator control loads. Higher harmonic terms of servo flap deflection were found to be most pronounced in flatwise bending moment, transmission vertical vibration and pitch link vibratory load equations. The existing test hardware represents a satisfactory configuration for demonstrating MCTR technology and defining a data base for additional wind tunnel testing. T.M.

Controlled Terms: AERODYNAMIC CHARACTERISTICS / AERODYNAMIC STABILITY / *BENDING MOMENTS / BENDING VIBRATION / *CONTROLLABILITY / HELICOPTER PERFORMANCE / LOADING MOMENTS / *OPTIMIZATION / REGRESSION ANALYSIS / *ROTARY WINGS / *ROTOR AERODYNAMICS / *WIND TUNNEL TESTS

Reference no 134

81A17517 NASA ISSUE 5 Category 7

Engine-airframe transient compatibility - Analysis and test

ALWANG, J. R.

800000 In: Specialists' Meeting on Helicopter Propulsion Systems, Williamsburg, Va., November 6-8, 1979, Technical Papers. (A81-17501 05-01) Washington, D.C., American Helicopter Society, 1980. 9 p. AA(Boeing Vertol Co., Philadelphia, Pa.) 9 p. Jpn. 654

This paper discusses dynamic analysis and test requirements related to engine-airframe compatibility in two areas: drive system torsional stability and transient response. Prediction of stability characteristics using linear techniques and correlation with experimental data are reviewed. Control testing, leading to determination of worst-case conditions, is presented with an aircraft flight test procedure for providing an accurate evaluation of propulsion system stability. The importance of developing a nonlinear component-by-component engine simulation for optimizing control functional configurations and transient schedules is emphasized. Use of a dynamic simulation based on steady state matching deck performance for analyzing specific interface problems, including transient loss in compressor surge margin, is demonstrated. (Author)

Controlled Terms: *AIRFRAMES / COMPATIBILITY / DYNAMIC MODELS / *DYNAMIC STABILITY / ENGINE PARTS / *FLIGHT TESTS / *HELICOPTER ENGINES / LINEAR EQUATIONS / OPTIMAL CONTROL / *RIGID ROTOR HELICOPTERS / TORSIONAL VIBRATION / TRANSIENT RESPONSE / *VIBRATION EFFECTS

Reference no 135

81A17467 NASA ISSUE 5 Category 8

Problems of flight mechanics involved in all-weather helicopter operation

Flugmechanische Probleme bei der Realisierung des^Allwettereinsatzes von Hubschraubern

HAMEL, P.; GMELIN, B.

801200 Zeitschrift fuer Flugwissenschaften und Weltraumforschung, vol. 4, Nov.-Dec. 1980, p. 335-345. In German. AB(Deutsche Forschungs- und Versuchsanstalt fuer Luft- und Raumfahrt, Institut fuer Flugmechanik, Braunschweig, West Germany) 11 p. Refs. 12 Jpn. 656

In view of the inferior flight mechanical behavior of the helicopter, in particular, with respect to military all-weather requirements, some major factors influencing the flight mechanical behavior of the pilot/helicopter system are examined. Flight mechanical problem areas are elucidated, and the basic flight properties of the helicopter that would enable all-weather operation are discussed. Methods of optimizing basic helicopter flight characteristics are proposed. V.P.

Controlled Terms: *ALL-WEATHER AIR NAVIGATION / FLIGHT MECHANICS / *FLIGHT OPERATIONS / FLIGHT OPTIMIZATION / *HELICOPTER PERFORMANCE / MAN MACHINE SYSTEMS / NAP-OF-THE-EARTH NAVIGATION / *OPERATIONAL PROBLEMS

Reference no 136

81N16046# NASA ISSUE 7 Category 5

Preliminary Airworthiness Evaluation (PAE 1) of the YCH-47D helicopter

Final Report, Sep. - Dec. 1979

WILSON, G. W.; ADAM, C. F.; ARTHUR, S. F.; NIEMANN, J. R.; BOWERS, F. J.

Army Aviation Engineering Flight Activity, Edwards AFB, Calif. (AZ102498)

AD-A092633; USAAEFA-79-06 800500 150 p. Jpn. 858 HC A07/MF A01

The Preliminary Airworthiness Evaluation of the YCH-47D helicopter was conducted between 11 September and 6 December 1979. Seventeen flights were required for a total of 41.4 hours, of which 32.8 hours were productive. Testing was conducted at the Boeing Flight Test Facility at Wilmington, Delaware. The YCH-47D exhibits improved lift capability at a hover (both in and out of ground effect) when compared to the CH-47C with fiberglass rotor blades and T55-L-712 engines. Higher airspeeds are also possible at high gross weights. The AFCS was found to be an enhancing characteristic. Twenty shortcomings and one deficiency were also documented during the test. The deficiency of delay in power steering activation should be corrected prior to operational deployment. The noise level in the cockpit and cabin showed no apparent improvement over previous models and was of sufficient magnitude to induce temporary hearing loss without adequate protection. The vibration levels became excessive above 145 KIAS, increasing to unacceptable prior to V sub H, with 6 per rev vibrations being predominant. GRA

Controlled Terms: *AIRCRAFT STABILITY / *AVIONICS / *CH-47 HELICOPTER / *CONTROLLABILITY / *FLIGHT TESTS / HELICOPTER DESIGN / *HELICOPTER PERFORMANCE / HELICOPTER PROPELLER DRIVE / MANEUVERABILITY / ROTARY WINGS

Reference no 137

81N15007# NASA ISSUE 6 Category 8

A pure direct force/moment control for coaxial counterrotating rotors

WADIA, A. R.; FAIRCHILD, J. E.

Texas Univ. at Arlington. (TV553653)

800000 Dept. of Aerospace Engineering. AA(AiResearch Mfg. Co., Phoenix, Ariz.) 19 p. Jpn. 712 HC A02/MF A01

A simple first harmonic analysis using Fourier series is presented for the control of rigid coaxial and counterrotating helicopter rotors in hover. It it shown that by a particular combination of cyclic controls on each rotor, pure moments or forces can be generated in any direction. Such controls could give the helicopter maneuvering capabilities not possible in conventional machines. An elementary cockpit control configuration is suggested to implement the control strategy. E.D.K.

Controlled Terms: CYCLIC LOADS / FORCE DISTRIBUTION / *FOURIER SERIES / *HARMONIC ANALYSIS / *HELICOPTER CONTROL / *HOVERING STABILITY / *MANEUVERABILITY / MOMENT DISTRIBUTION / *ROTARY WINGS

Reference no 138

81N13051 NASA ISSUE 4 Category 5

Dynamics of a helicopter-slung load system

Ph.D. Thesis

SAMPATH, P.

Maryland Univ., College Park. (MI915766)

800000 164 p. Jpn. 437 Avail: Univ. Microfilms Order No. 8027137

Stability of a tandem rotor helicopter carrying a slung cargo container was investigated. Lagrange equations were used to write the equations of motion. The cables of the sling were modeled as massless rigid extensible rods, which collapse under compressive loads. Extensibility was provided by considering the rods as linear springs with viscous damping. Aerodynamics of the cable were neglected. Tabulated static aerodynamic data were considered for the helicopter as well as the load. The equations were divided into two sets, one representing the towing vehicle (referred to as Subsystem 1) and the other representing the slung load (referred to as Subsystem 2). Subsystem 2 corresponds to a wind tunnel model of a slung load. Stability of Subsystem 2 was investigated for various flow velocities and sling configurations. The results were compared with the results of wind tunnel tests of scaled models. The influence of the load oscillations on the helicopter is substantially reduced when the helicopter is flown with its stability augmentation system on. Without the intervention of the pilot, load oscillations cause the helicopter, with or without the stability augmentation system, to oscillate. Pilot inputs can also induce oscillations in the load. Dissert. Abstr.

Controlled Terms: AERODYNAMIC LOADS / *AERODYNAMIC STABILITY / *CARGO / *EQUATIONS OF MOTION / *FLOW VELOCITY / *HELICOPTER CONTROL / *OSCILLATIONS / SCALE MODELS / WIND TUNNEL TESTS

Reference no 139

81N13050# NASA ISSUE 4 Category 5

Helicopter In-flight Validation System (HELIVALS)

Final Report

FREZELL, T. L.; HERALD, G.; CAMDEN, R. S.; FRY, C. A.

Human Engineering Labs., Aberdeen Proving Ground, Md. (H6521544)

AD-A091129; HEL-TM-17-80 800800 25 p. Jpn. 437 HC A02/MF A01

This report described the Helicopter In-Flight Validation System (HELIVALS). The system monitors all six degrees of freedom of the helicopter flight control position to include cyclic, collective and anti-torque pedals; along with airspeed, altitude (both barometric and absolute), and geographic position. These data are recorded digitally on a magnetic tape recorder mounted in the helicopter. The recorded data is then processed and reduced in a ground station. GRA

Controlled Terms: *AIRCRAFT MANEUVERS / *DATA ACQUISITION / DATA PROCESSING / *DEGREES OF FREEDOM / FLIGHT CHARACTERISTICS / FLIGHT PATHS / *FLIGHT RECORDERS / *HELICOPTER PERFORMANCE / MANEUVERABILITY / PILOT PERFORMANCE / QUALITY CONTROL

Reference no 140

81N12071# NASA ISSUE 3 Category 5

Stability of nonuniform rotor blades in hover using a mixed formulation

STEPHENS, W. B.; HODGES, D. H.; AVILA, J. H.; KUNG, R. M.

Army Research and Technology Labs., Moffett Field, Calif. (A2025071)

AD-A090756 800000 Aeromechanics Lab. Presented at the 6th European Rotorcraft and Powered Lift Aircraft Forum, Bristol, England, 16-19 Sep. 1980 AC(Technology Development of California, Santa Clara) AD(Technology Development of California, Santa Clara) 20 p. Jpn. 301 HC A02/MF A01

A mixed formulation for calculating static equilibrium and stability eigenvalues of nonuniform rotor blades in hover is presented. The static equilibrium equations are nonlinear and are solved by an accurate and efficient collocation method. The linearized perturbation equations are solved by a one step, second-order integration scheme. The numerical results correlate very well with published results from a nearly identical stability analysis based on a displacement formulation. Slight differences in the results are traced to terms in the equations that relate moments to to derivatives of rotations. With the present ordering scheme, in which terms of the order of squares of rotations are neglected with respect to unity, it is not possible to achieve completely equivalent models based on mixed and displacement formulations. A study of the one step methods reveals that a second order Taylor expansion is necessary to achieve good convergence for nonuniform rotating blades. Numerical results for a hypothetical nonuniform blade, including the nonlinear static equilibrium solution, were obtained with no more effort or computer time than that required for a uniform blade with the present analysis. GRA

Controlled Terms: *AERODYNAMIC LOADS / BOUNDARY VALUE PROBLEMS / COMPUTERIZED SIMULATION / *FINITE ELEMENT METHOD / *HELICOPTER PERFORMANCE / HELICOPTER PROPELLER DRIVE / *HOVERING STABILITY / *ROTARY WINGS / TABLES (DATA) / WING LOADING

Reference no 141

81N11033# NASA ISSUE 2 Category 5

Experimental and analytical studies of a model helicopter rotor in hover

CARADONNA, F. X.; TUNG, C.

Army Research and Technology Labs., Moffett Field, Calif. (A2025071)

AD-A089780; REPT-25 800000 Aeromechanics Lab. Presented at the European Rotorcraft and Powered Lift Aircraft Forum, Bristol, England, 16-19 Sep. 1980 20 p. Jpn. 155 HC A02/MF A01

The present study is a benchmark test to aid the development of various rotor performance codes. The study involves simultaneous blade pressure measurements and tip vortex surveys. Measurements were made for a wide range of tip Mach numbers including the transonic flow regime. The measured tip vortex strength and geometry permit effective blade loading predictions when used as input to a prescribed wake lifting surface code. It is also shown that with proper inflow and boundary layer modeling, the supercritical flow regime can be accurately predicted. GRA

Controlled Terms: *HELICOPTER PERFORMANCE / HELICOPTER WAKES / HOT WIRE ANEMOMETERS / *HOVERING / PREDICTION ANALYSIS TECHNIQUES / PRESSURE DISTRIBUTION / PRESSURE SENSORS / *ROTARY WINGS / *ROTOR AERODYNAMICS / TRANSONIC FLOW / WING TIP VORTICES

Reference no 142

81A10768 NASA ISSUE 1 Category 5

Helicopter tail configurations to survive tail rotor loss

BROCKLEHURST, A.; TAYLOR, P.

800000 Vertica, vol. 4, no. 2-4, 1980, p. 107-119. Research supported by the Westland Helicopters. AA(Westland Helicopters, Ltd., Yeovil, Somerset, England) AB(Southampton, University, Southampton, England) 13 p. Refs. 5 Jpn. 3

Recently the US Army have specified that a helicopter must be capable of returning from its mission after suffering a tail rotor loss. The helicopter should possess sufficient directional stability to fly at the minimum power speed with a sideslip angle of not more than 20 deg. A simple theory, describing the yawing oscillation of a helicopter, has been applied to a typical helicopter in order to identify the stability implications on the aerodynamic design of meeting the above tail rotor loss criterion. The fin area required, for a fin and single tail rotor configuration, to meet both the above criterion and to ensure adequate lateral stability characteristics was large even if camber and incidence were used. The same helicopter but with twin tail rotors and no fin was investigated. This configuration has additional advantages including the unique ability to land in confined places after the loss of a tail rotor. (Author)

Controlled Terms: *AERODYNAMIC CHARACTERISTICS / *AIRCRAFT ACCIDENTS / *AIRCRAFT SURVIVABILITY / *DIRECTIONAL STABILITY / EMERGENCIES / EQUIPMENT SPECIFICATIONS / FINS / *HELICOPTER TAIL ROTORS / *MILITARY HELICOPTERS / TAIL SURFACES / YAW

Reference no 143

81A10767 NASA ISSUE 1 Category 8

Piloted simulation studies of helicopter agility

TOMLINSON, B. N.; PADFIELD, G. D.

800000 Vertica, vol. 4, no. 2-4, 1980, p. 79-106. AB(Royal Aircraft Establishment, Bedford, England) 28 p. Refs. 17 Jpn. 4

The need for helicopters to operate close to the ground and near obstacles has prompted a critical look at design features which affect performance and handling qualities in this environment. Some experiments using a ground-based flight simulator have been conducted to investigate this subject and to obtain data on helicopter agility. These experiments required the development of a general mathematical model capable of representing helicopter flight, including gross maneuvers, from hover to cruise and validation by comparison with flight tests. An exacting low level flying course was created on a model ground terrain and formed the primary task for the six pilots involved in the experiments. A set of rotors were represented which differed in blade flapping stiffness and inertia (Lock number). The paper describes these aspects and then goes on to describe how the simulated helicopter was flown over the agility course with each rotor

to investigate the effects of rotor design. Some of the theoretical consequences of these variations are outlined and the results of piloted flights in the simulator described. (Author)

Controlled Terms: *AIRCRAFT MANEUVERS / AIRSPEED / *COCKPIT SIMULATORS / *CONTROLLABILITY / FLIGHT PATHS / *FLIGHT SIMULATION / *HELICOPTER PERFORMANCE / LOW ALTITUDE / MATHEMATICAL MODELS / PILOT PERFORMANCE

Reference no 144

81N10703# NASA ISSUE 1 Category 53

A study on pilot/observer interaction in aligning a helicopter with a target

DANNEBERG, E.

European Space Agency, Paris (France). (E6854803)

ESA-TT-480; DFVLR-FB-79-04 800100 Transl. into ENGLISH of ""Untersuchung des Zusammenwirkens von Pilot und Beobachter beim Ausrichten des Hubschraubers auf ein Ziel'', Rept. DFVLR-FB-79-04, DFVLR, Brunswick, Nov. 1978 Original report in GERMAN previously announced as N79-30945 32 p. Jpn. 107 HC A03/MF A01; DFVLR, Cologne DM 8,70

Speed and stability of the alignment of a helicopter with a target are investigated by means of a system simulation. Variables are the maximum yaw rate of the helicopter, the maximum pan rate of the steerable sight or sensor, and their optical magnification. The results obtained provided favorable combinations of these parameters for two different cases, i.e., target acquisition and target already picked up by means of the sight or sensor, respectively. Author (ESA)

Controlled Terms: *AIRCRAFT MANEUVERS / ATTACK AIRCRAFT / DETECTION / DIRECTIONAL STABILITY / FLIGHT SIMULATION / *MILITARY HELICOPTERS / OPTICAL MEASURING INSTRUMENTS / OPTICAL TRACKING / *TARGET ACQUISITION / YAW

Reference no 145

80N33777# NASA ISSUE 24 Category 39

Stability of nonuniform rotor blades in hover using a mixed formulation

STEPHENS, W. B.; HODGES, D. H.; AVILA, J. H.; KUNG, R. M.

National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)

NASA-TM-81226; A-8314; AVRADCOM-TR-80-A-10; PAPER-13 800800 Presented at the 6th European Rotorcraft and Powered Lift Aircraft Forum, Bristol, England, 16-19 Sep. 1980 AA(Army Research and Technology Labs., Moffett Field, Calif.) AB(Army Rsearch and Technology Labs., Moffett Field, Calif.) AC(Technology Development of California, Santa Clara) AD(Technology Development of California, Santa Clara) 23 p. Jpn. 3288 HC A02/MF A01

A mixed formulation for calculating static equilibrium and stability eigenvalues of nonuniform rotor blades in hover is presented. The static equilibrium equations are nonlinear and are solved by an accurate and efficient collocation method. The linearized perturbation equations are solved by a one step, second order integration scheme. The numerical results correlate very well with published results from a nearly identical stability analysis based on a displacement formulation. Slight differences in the results are traced to terms in the equations that relate moments to derivatives of rotations. With the present ordering scheme, in which terms of the order of squares of rotations are neglected with respect to unity, it is not possible to achieve completely equivalent models based on mixed and displacement formulations. The one step methods reveal that a second order Taylor expansion is necessary to achieve good convergence for nonuniform rotating blades. Numerical results for a hypothetical nonuniform blade, including the nonlinear static equilibrium solution, were obtained with no more effort or computer time than that required for a uniform blade. Author

Controlled Terms: COMPUTER TECHNIQUES / EIGENVALUES / *HOVERING / NONLINEARITY / NUMERICAL ANALYSIS / *ROTARY WINGS / *STATIC STABILITY

Reference no 146

80N29370# NASA ISSUE 20 Category 8

Results of a simulator investigation of control system and display variations for an attack helicopter mission

AIKEN, E. W.; MERRILL, R. K.

Army Research and Technology Labs., Moffett Field, Calif. (A2025071)

AD-A085812 800500 Aeromechanics Lab. Presented at 36th Ann. Natl. Forum of the AHS, Washington, D.C., May, 1980 25 p. Jpn. 2660 HC A02/MF A01

A piloted simulator experiment designed to a sess the effects on overall system performance and pilot workload of variations in control system characteristics and display format and logic for a nighttime attack helicopter mission is described. The simulation facility provided a representation of a helmet-mounted display image consisting of flight-control and fire-control symbology superimposed on the background video from a simulated forward-looking infrared sensor. Control systems ranging from the baseline stability and control augmentation system to various hover augmentation schemes were investigated together with variations in the format and logic of the superimposed symbology. Selected control system and display failures were also simulated. The results of the experiment indicate that the baseline control/display system is unsatisfactory without improvement for the evaluation task which included a hovering target search and acquisition. Significant improvements in pilot rating were achieved by both control system and display variations. GRA

Controlled Terms: *AIRCRAFT MANEUVERS / *ATTACK AIRCRAFT / *FIRE CONTROL / *FLIGHT CONTROL / FLIR DETECTORS / HELMET MOUNTED DISPLAYS / *MAN MACHINE SYSTEMS / *MILITARY HELICOPTERS / NIGHT FLIGHTS (AIRCRAFT) / PILOT PERFORMANCE / TARGET ACQUISITION

Reference no 147

80N29294# NASA ISSUE 20 Category 5

An experimental investigation of the effects of aeroelastic couplings on aeromechanical stability of a hingeless rotor helicopter

BOUSMAN, W. G.

Army Research and Technology Labs., Moffett Field, Calif. (A2025071)
AD-A085819 800000 Aeromechanics Lab. Presented at the Ann. Forum of the Am. Helicopter Soc., Washington, D.C., May 1980 14 p. Jpn. 2650 HC A02/MF A01

A 1.62 m diameter rotor model was used to investigate aeromechanical stability, and the results were compared to theory. Configurations tested included: (1) a nonmatched stiffness rotor as a baseline, (2) the baseline rotor with negative pitch-lag coupling, (3) the combination of negative pitch-lag and structural flap-lag coupling on the baseline rotor, (4) a matched stiffness rotor, and (5) a matched stiffness rotor with negative pitch-lag coupling. The measured lead-lag regressing mode damping of the five configurations agreed well with theory, but only the matched stiffness case with negative pitch-lag coupling was able to stabilize the air resonance mode. Comparison of theory and experiment for the damping of the body modes showed significant differences that may be related to rotor inflow dynamics. GRA

Controlled Terms: *AERODYNAMIC STABILITY / AEROELASTICITY / EXPERIMENTAL DESIGN / *HELICOPTERS / PITCH (INCLINATION) / *RIGID ROTORS

Reference no 148
80N29288# NASA ISSUE 20 Category 5
Wessex helicopter/sonar dynamics study ARL program description and operation
WILLIAMS, N. V.; GUY, C. R.; GILBERT, N. E.
Aeronautical Research Labs., Melbourne (Australia). (AF441057)
ARL-AERO-NOTE-385; AR-001-603 790200 38 p. Jpn. 2649 HC A03/MF A01

A computer program, representing the dynamic behavior in flight of the Wessex MK.31B helicopter and its antisubmarine warfar (ASW) sonar equipment, is described. The program is intended for operation on a PDP 10 computer and is written in CSMP 10 (ARL) simulation language. Instructions for setting up a particular simulated flight maneuver are given, together with details of program verification. Author

Controlled Terms: AERODYNAMICS / AUTOMATIC FLIGHT CONTROL / *COMPUTER PROGRAMS / COMPUTERIZED SIMULATION / *DYNAMIC CHARACTERISTICS / *FLIGHT MECHANICS / *FLIGHT SIMULATION / *HELICOPTERS / *SONAR / TRANSDUCERS

Reference no 149
80N28369# NASA ISSUE 19 Category 8
Analytical design and evaluation of an active control system for helicopter vibration reduction and gust response alleviation
TAYLOR, R. B.; ZWICKE, P. E.; GOLD, P.; MIAO, W.
United Technologies Research Center, East Hartford, Conn. (UW324219)
NASA-CR-152377 800700 NAS2-10121 Prepared in cooperation with Sikorsky Aircraft, Stratford, Conn. 165 p. Jpn. 2524 HC A08/MF A01

An analytical study was conducted to define the basic configuration of an active control system for helicopter vibration and gust response alleviation. The study culminated in a control system design which has two separate systems: narrow band loop for vibration reduction and wider band loop for gust response alleviation. The narrow band vibration loop utilizes the standard swashplate control configuration to input controller for the vibration loop is based on adaptive optimal control theory and is designed to adapt to any flight condition including maneuvers and transients. The prime characteristics of the vibration control system is its real time capability. The gust alleviation control system studied consists of optimal sampled data feedback gains together with an optimal one-step-ahead prediction. The prediction permits the estimation of the gust disturbance which can then be used to minimize the gust effects on the helicopter. E.D.K.

Controlled Terms: *ACTIVE CONTROL / AIRBORNE/SPACEBORNE COMPUTERS / *GUST ALLEVIATORS / *HELICOPTER CONTROL / REAL TIME OPERATION / *ROTOR AERODYNAMICS / *VIBRATION DAMPING / *WIND EFFECTS

Reference no 150
80N28298# NASA ISSUE 19 Category 1
A comprehensive analytical model of rotorcraft aerodynamics and dynamics. Part 3: Program manual
JOHNSON, W.
National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)
NASA-TM-81184; AVRADCOM-TR-80-A-7; A-8102 800600 Prepared in cooperation with Army Aviation Research and Development Command, St. Louis, Mo. 155 p. Jpn. 2513 HC A08/MF A01

The computer program for a comprehensive analytical model of rotorcraft aerodynamics and dynamics is described. This analysis is designed to calculate rotor performance, loads, and noise; the helicopter vibration and gust response; the flight dynamics and handling qualities; and the system aeroelastic stability. The analysis is a combination of structural, inertial, and aerodynamic models that is applicable to a wide range of problems and a wide class of vehicles. The analysis is intended for use in the design, testing, and evaluation of rotors and rotorcraft and to be a basis for further development of rotary wing theories. Author

Controlled Terms: *AERODYNAMIC LOADS / *AIRCRAFT NOISE / *COMPUTER PROGRAMS / COMPUTERIZED SIMULATION / *HELICOPTER CONTROL / *HELICOPTER PERFORMANCE / ROTARY WING AIRCRAFT / *ROTOR AERODYNAMICS / *VIBRATION

Reference no 151
80N24329# NASA ISSUE 15 Category 8
Flight evaluation of nondimensional static longitudinal stability test methods
Final Report, Nov. 1974 - Feb. 1975
FERRELL, K. R.; BOIRUN, B. H.; JENKS, J. E., JR.
Army Aviation Engineering Flight Activity, Edwards AFB, Calif. (AZ102498)
AD-A082831; USAAEFA-74-87 790700 66 p. Jpn. 1942 HC A04/MF A01

Stability and control testing of helicopters at this activity has included performing discrete tests at representative mission

conditions, as well as extremes of altitude, airspeed, vertical speed, rotor speed, gross weight, and center of gravity location. Special studies and tests concluded that the use of nondimensional parameters commonly used in performance analysis such as the thrust coefficient, advance ratio, and advancing tip Mach number could be used to reduce the number of independent test variables in stability and control testing. These studies also indicated that constant referred rotor speed might be better for isolation of compressibility effects than the conventional constant rotor speed method for obtaining nondimensional test data. This test investigated the relative advantage of obtaining static longitudinal stability and trim control data as a function of the nondimensional parameters using both the constant referred and constant rotor speed methods on a UH-1M helicopter. Within the accuracy of the test instrumentation, the static longitudinal trim control position data were found to be a function of thrust coefficient and advance ratio. The collective fixed static longitudinal stability was primarily a function of advance ratio only. The advancing tip Mach number did not have a significant effect on the longitudinal stability data on this test helicopter. Static longitudinal stability and control trim position can be defined by obtaining data over the full range of the thrust coefficient and advance ratio envelope and then using the nondimensional method to calculate intermediate value. GRA

Controlled Terms: AIRSPEED / COMPRESSIBILITY EFFECTS / *FLIGHT STABILITY TESTS / HELICOPTER CONTROL / *HELICOPTER PERFORMANCE / INDEPENDENT VARIABLES / *LONGITUDINAL STABILITY / ROTOR SPEED / *STATIC STABILITY / THRUST / UH-1 HELICOPTER

Reference no 152

80N22357# NASA ISSUE 13 Category 8

The design, testing and evaluation of the MIT individual-blade-control system as applied to gust alleviation for helicopters

Final Report

MCKILLIP, R. M., JR.

Massachusetts Inst. of Tech., Cambridge. (MJ700802)

NASA-CR-152352; ASRL-TR-196-1 800200 NSG-2266 Aeroelastic and Structures Research Lab. 92 p. Jpn. 1672 HC A05/MF A01

A type of active control for helicopters was designed and tested on a four foot diameter model rotor. A single blade was individually controlled in pitch in the rotating frame over a wide range of frequencies by electromechanical means. By utilizing a tip mounted accelerometer as a sensor in the feedback path, significant reductions in blade flapping response to gust were achieved at the gust excitation frequency as well as at super and subharmonics of rotor speed. E.D.K.

Controlled Terms: ACCELEROMETERS / EQUATIONS OF MOTION / *GUST ALLEVIATORS / HARMONICS / *HELICOPTER CONTROL / *LONGITUDINAL CONTROL / *ROTARY WINGS / *ROTOR AERODYNAMICS / ROTOR SPEED / *SERVOMECHANISMS / SYSTEM EFFECTIVENESS / WIND TUNNEL TESTS

Reference no 153

80N22310# NASA ISSUE 13 Category 5

Improved maneuver criteria evaluation program --- flight path equations of motion for the h-58 and Oh-58a aircraft

Final report, Sep. 1976 - Jul 1979

WOOD, T.; WAAK, T.

Textron Bell Helicopter, Fort Worth, Tex. (TV739419)

AD-A080408; USARTL-TR-79-20 791100 DAAJ02-76-C-0064; DA PROJ. 1F2-63211-D-157 79 p. Jpn. 1666 HC A05/MF A01

The Maneuver Criteria Evaluation Program (MCEP) is a digital computer program that solves the flight path equation of motion for a helicopter without auxiliary propulsion. The use of basic work, energy, and power relationships makes possible accurate representation of flight path trajectories. MCEP can be used to aid in the development of maneuver requirements that provide the necessary maneuver capability to perform the desired mission. The desired mission is simulated in MCEP by using individual flight controllers to 'fly' the helicopter through the mission profile. Key maneuver parameters are monitored throughout the flight profile to provide insight into the performance of the helicopter in achieving the desired flight trajectory. Three maneuvers have been modified to allow rotor rpm to be bled to use some of the rotor's stored energy. These maneuvers are a constant altitude acceleration maneuver, a collective pop-up maneuver, and a sideward acceleration maneuver. Correlation with flight test data is established to validate the bleed rpm maneuvers. The appendix to the report, the User's Guide, contains the detailed information necessary for setting up an input data deck for MCEP. GRA

Controlled Terms: *AIRCRAFT MANEUVERS / *COMPUTER PROGRAMS / DIGITAL SIMULATION / EQUATIONS OF MOTION / *FLIGHT PATHS / GROUND EFFECT / *HELICOPTER PERFORMANCE / *OH-58 HELICOPTER / ROTARY WINGS / ROTOR AERODYNAMICS / TRAJECTORIES

Reference no 154

80N20265# NASA ISSUE 11 Category 5

Preliminary airworthiness evaluation, AH-1G with the airborne target acquisition fire control system and the hellfire modular missile system installed

Final Report, Jun. 1978 - Jan. 1979

MOE, P. J.; SMITH, R. B.

Army Aviation Engineering Flight Activity, Edwards AFB, Calif. (AZ102498)

AD-A078340; USAAEFA-78-02 790700 43 p. Jpn. 1377 HC A03/MF A01

The AH-1G helicopter with the HELLFIRE modular Missile System (HMMS) and the Airborne Target Acquisition Fire Control System (ATAFCS) is being used as a surrogate trainer for the YAH-64 helicopter. The United States Army Aviation Engineering Flight Activity was tasked to provide quantitative and qualitative data on the handling qualities of the helicopter, obtain limited level flight performance data, and obtain limited handling qualities of the helicopter with only the

ATAFCS installed. The test helicopter was a production AH-1G helicopter (212 tail rotor) modified with an ATAFCS mockup and carrying eight HELLFIRE missiles. Six productive flight test hours were flown in six flights. No shortcomings or deficiencies attributable to HMMS and ATAFCS installation were found. The AH-1G helicopter, with HMMS and ATAFCS installed, exhibits an additional equivalent flat plate area of 4.0 ft2 compared to the standard AH-1G helicopter. The handling qualities of the helicopter with only the ATAFCS installed are essentially the same as the production AH-1G helicopter. GRA

Controlled Terms: *AH-1G HELICOPTER / *AIRCRAFT RELIABILITY / AIRCRAFT STABILITY / *CONTROLLABILITY / FIRE CONTROL / FLIGHT CHARACTERISTICS / *FLIGHT TESTS / HELICOPTER DESIGN / *HELICOPTER PERFORMANCE / MISSILE SYSTEMS / TARGET ACQUISITION

Reference no 155

80N19101# NASA ISSUE 10 Category 5

Rotorcraft identification experience

KALETKA, J.

Deutsche Forschungs- und Versuchsanstalt fuer Luft- und Raumfahrt, Brunswick (West Germany). (DO696666)

791100 Inst. fuer Flugmechanik. In AGARD Parameter Identification 32 p (SEE N80-19094 10-05) 32 p. Jpn. 1240 HC A16/MF A01

An overview of the identification of stability and control derivatives of the rotorcraft with respect to practical aspects and applications is presented. First an introduction to the basic dynamics and control of helicopters is given. The helicopter characteristics causing difficulties in the identification are discussed. Measurement and sensor problems are also discussed. Approaches to overcome the difficulties are presented. Emphasis is placed on the following two key elements of the identification procedure: (1) the selection of adequate mathematical models and identifiable derivatives of the helicopter to isolate significant model effects; and (2) possibilities of increasing the information content of flight test data by appropriate system excitation and by multiple-run evaluations. Identification results obtained from simulated and flight test data of helicopters by applying different identification methods are presented. R.E.S.

Controlled Terms: *AERODYNAMIC COEFFICIENTS / *AIRCRAFT CONTROL / *AIRCRAFT STABILITY / DATA ACQUISITION / FLIGHT CHARACTERISTICS / HELICOPTERS / MATHEMATICAL MODELS / *ROTARY WING AIRCRAFT / SYSTEMS ANALYSIS

Reference no 156

80N18050# NASA ISSUE 9 Category 8

Rotors in forward flight and dynamic stall

pale reculante et decrochage dynamique

RENAUD, J.

Aerospatiale Usines de Toulouse (France). (AG944762)

AAAF-NT-79-20; ISBN-2-7170-0547-1 790000 Helicopters Div. Presented at 15th Colloq. d'Aerodyn. Appl., Marseille, 7-9 Nov. 1978 26 p. Paris Assoc. Aeron. et Astronique de France Jpn. 1095 HC A03/MF A01; CEDOCAR, Paris FF 17 (France and EEC) FF 21 (others)

Dynamic stall of rotors in forward flight is studied in terms of the limiting on helicopter performance. Manifestations of the phenomena depend on the characteristics of the rotors and on operating conditions. The lift of the rotor, forward velocity, engine speed, and the interaction of vortices with the blade surface all contribute to the development of stall. Experiments demonstrating the unsteady nature of the phenomena are reported. Simulation of dynamic stall phenomena on an oscillating model shows that the delay in unsteady stalling is associated with the development of a system of vortices along the leading edge. A two dimensional analysis is used in modeling dynamic stall. Author (ESA)

Controlled Terms: *AERODYNAMIC STALLING / FLIGHT SIMULATION / HELICOPTER PERFORMANCE / *ROTARY WINGS / *ROTATING STALLS / ROTOR LIFT / ROTOR SPEED / WIND TUNNEL MODELS

Reference no 157

80N14049# NASA ISSUE 5 Category 2

Effect of tip planform on blade loading characteristics for a two-bladed rotor in hover

BALLARD, J. D.; ORLOFF, K. L.; LUEBS, A. B.

National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)

NASA-TM-78615; A-7939 791100 AC(Gates Lear Corp., Wichita, Kan.) 89 p. Jpn. 557 HC A05/MF A01

A laser velocimeter was used to study the flow surrounding a 2.13 m diam. two-bladed, teetering model-scale helicopter rotor operating in the hover condition. The rotor system employed interchangeable blade tips over the outer 25% radius. A conventional rectangular planform and an experimental ogee tip shape were studied. The radial distribution of the blade circulation was obtained by measuring the velocity tangent to a closed rectangular contour around the airfoil section at a number of radial locations. A relationship between local circulation and bound vorticity was invoked to obtain the radial variations in the sectional lifting properties of the blade. The tip vortex-induced velocity was also measured immediately behind the generating blade and immediately before the encounter with the following blade. The mutual influence between blade loading, shed vorticity, and the structure of the encountered vortex are quantified by the results presented and are discussed comparatively for the rectangular and ogee planforms. The experimental loading for the rectangular tip is also compared with predictions of existing rotor analysis. Author

Controlled Terms: AERODYNAMIC LOADS / *BLADE TIPS / *FLOW MEASUREMENT / *HOVERING / LASER DOPPLER VELOCIMETERS / OGEE SHAPE / *PLANFORMS / RADIAL DISTRIBUTION / *ROTARY WINGS / *ROTOR AERODYNAMICS / SCALE MODELS / *VELOCITY DISTRIBUTION / VORTICES

Reference no 158

80N12088# NASA ISSUE 3 Category 5

AH-1G lateral flight performance test

HEPLER, L. J.; KISHI, J. S.; WINN, A. L.

Army Aviation Test Activity, Edwards AFB, Calif. (AZ168112)

AD-A072868; USAASTA-71-43 790630 21 p. Jpn. 289 HC A02/MF A01

The lateral flight performance of the tractor-tail-rotor-configured AH-1G helicopter was evaluated at gross weights of 8500 and 9500 pounds. The AH-1G is capable of 0.53 g and 0.38 g accelerations in right and left lateral flight, respectively, at 8500 pounds. At a 9500 pound gross weight, acceleration was 0.23 g to the right and 0.20 g to the left. Aircraft handling qualities and time required to attain maximum lateral velocity are dependent upon rate and type of control application. A step-type lateral control input permitted rapid attainment of accelerating attitude but produced yaw oscillations which resulted in loss of directional control and caused high pilot workload in stabilizing power and aircraft attitude. Ramp inputs delayed establishment of the accelerating attitude. A modified pulse input induced negligible roll and yaw oscillation and produced rapid establishment of the desired roll attitude. Direction-of-flight reversals induced large power transients that frequently exceeded the transmission torque limits. Precise heading and pitch attitude could not be maintained during the reversals. Airspeed in lateral flight could not be determined without special airspeed measuring devices. The sideward airspeed limit was exceeded frequently in left lateral flight with no warning or cue to the pilot that the limit had been reached. GRA

Controlled Terms: ACCELERATION (PHYSICS) / *AIRCRAFT MANEUVERS / AIRSPEED / ATTITUDE CONTROL / *LATERAL STABILITY / MANUAL CONTROL / *MILITARY HELICOPTERS / *PERFORMANCE TESTS

Reference no 159

80A45902 NASA ISSUE 20 Category 8

Helicopter stability and control test methodology

BLAKE, B. B.; LUNN, K.

AIAA 80-1610 800000 In: Atmospheric Flight Mechanics Conference, Danvers, Mass., August 11-13, 1980, Collection of Technical Papers. (A80-45855 20-01) New York, American Institute of Aeronautics and Astronautics, Inc., 1980, p. 456-464. AB(Boeing Vertol Co., Philadelphia, Pa.) 9 p.

This paper reviews the techniques employed at Boeing Vertol to predict helicopter behavior for the frequency spectrum progressing from quasi-static through piloted flight dynamics to rotor order modes. Material covered includes flying qualities and aeromechanical stability testing of a Bearingless Main Rotor helicopter. Techniques for in-flight determination of rotor modal damping and progress on use of a NASA/Army developed parameter identification program are discussed. Prediction methods, build up techniques, and test monitoring for such potentially hazardous tests as height velocity determination and external sling load stability are examined. A forecast is made of advances to be expected in the near future. (Author)

Controlled Terms: AERODYNAMIC CHARACTERISTICS / *AIRCRAFT STABILITY / BEARINGLESS ROTORS / *CONTROL STABILITY / CONTROLLABILITY / FLIGHT HAZARDS / *FLIGHT STABILITY TESTS / *HELICOPTER CONTROL / IN-FLIGHT MONITORING / PARAMETER IDENTIFICATION / PERFORMANCE PREDICTION / VIBRATION DAMPING

Reference no 160

80A45887 NASA ISSUE 20 Category 8

A critique of handling qualities specifications for U.S. military helicopters

KEY, D. L.

AIAA 80-1592 800000 In: Atmospheric Flight Mechanics Conference, Danvers, Mass., August 11-13, 1980, Collection of Technical Papers. (A80-45855 20-01) New York, American Institute of Aeronautics and Astronautics, Inc., 1980, p. 305-319. AA(U.S. Army, Aeromechanics Laboratory, Moffett Field, Calif.) 15 p. Refs. 39

Inadequacies in the military specification for helicopter handling qualities, MIL-H-8501A, have long been recognized, and the latest procurements by the U.S. Army used special 'Prime Item Development Specifications' (PIDS). This paper assesses the efficacy of these 'PIDS' and suggests that changes should be made. In particular, the structure developed in MIL-F-8785B(ASG) (the specification for flying qualities of piloted airplanes) should be incorporated. Improved requirements must be based on a systematic database and concentrated on topics most important in preliminary design: static and dynamic stability, control power and sensitivity, and the interaction with controllers and displays. Emphasis should be on current military helicopter missions and helicopter idiosyncrasies such as cross-coupling, nonlinearities, and higher order dynamics. (Author)

Controlled Terms: AERODYNAMIC STABILITY / *AIRCRAFT SPECIFICATIONS / *CONTROLLABILITY / DATA BASES / *HELICOPTER CONTROL / HELICOPTER DESIGN / *MILITARY HELICOPTERS / STANDARDS / STATIC STABILITY

Reference no 161

80A45556 NASA ISSUE 19 Category 5

A new approach to active control of rotorcraft vibration

GUPTA, N. K.; DU VAL, R. W.; FULLER, J.

National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)

AIAA 80-1778 800000 In: Guidance and Control Conference, Danvers, Mass., August 11-13, 1980, Collection of Technical Papers. (A80-45514 19-17) New York, American Institute of Aeronautics and Astronautics, Inc., 1980, p. 347-358. AA(Integrated Systems, Inc., Stanford, Calif.) AB(NASA, Ames Research Center, Moffett Field, Calif.) AC(Systems Control, Inc., Palo Alto, Calif.) 12 p. Refs. 13 Jpn. 3454

A state-variable feedback approach is utilized for active control of rotorcraft vibration. Fuselage accelerations are passed through undamped second-order filters with resonant frequencies at N/rev. The resulting outputs contain predominantly the N/rev vibration components, phase shifted by 180 deg, and are used to drive the blade pitch to cancel this component of fuselage vibration. The linear-quadratic-gaussian (LQG) method is used to design a feedback control system utilizing these filtered accelerations. The design is based on a nine-degree-of-freedom linear model of the Rotor System Research Aircraft (RSRA) in hover and is evaluated on a nonlinear blade-element simulation of the RSRA for this flight condition. The system is shown to essentially eliminate vibrations at N/rev in all axes. The required blade-pitch amplitude is within the capability of conventional actuators at the N/rev frequency. (Author)

Controlled Terms: *ACCELERATION PROTECTION / *AIRCRAFT CONTROL / AIRCRAFT DESIGN / DEGREES OF FREEDOM / *FEEDBACK CONTROL / FLIGHT SIMULATION / FUSELAGES / PHASE SHIFT / RESONANT FREQUENCIES / *ROTORCRAFT AIRCRAFT / STATE VECTORS / *STRUCTURAL VIBRATION / *VIBRATION DAMPING

Reference no 162

80A45523 NASA ISSUE 19 Category 8

A model for helicopter guidance on spiral trajectories

MENDENHALL, S.; SLATER, G. L.

General Electric Co., Cincinnati, Ohio. (GG100602)

AIAA 80-1721 800000 NCA2-OR-130-801 In: Guidance and Control **Conference, Danvers, Mass.,** August 11-13, 1980, Collection of **Technical Papers.** (A80-45514 19-17) **New** York, American Institute of **Aeronautics and Astronautics,** Inc., 1980, p. 62-71. AA(General Electric Co., Cincinnati, Ohio) AB(Cincinnati, University, Cincinnati, Ohio) 10 p. Refs. 6 Jpn. 3457

A point mass model is developed for helicopter guidance on spiral trajectories. A fully coupled set of state equations is developed and perturbation equations suitable for 3-D and 4-D guidance are derived and shown to be amenable to conventional state variable feedback methods. Control variables are chosen to be the magnitude and orientation of the net rotor thrust. Using these variables reference controls for nonlevel accelerating trajectories are easily determined. The effects of constant wind are shown to require significant feedforward correction to some of the reference controls and to the time. Although not easily measured themselves, the controls variables chosen are shown to be easily related to the physical variables available in the cockpit. (Author)

Controlled Terms: *AIRCRAFT GUIDANCE / EQUATIONS OF MOTION / *FEEDBACK CONTROL / FLIGHT TIME / *HELICOPTER CONTROL / *MATHEMATICAL MODELS / SPIRALS / *STATE VECTORS / *TURNING FLIGHT / V/STOL AIRCRAFT / WIND EFFECTS

Reference no 163

80A35107 NASA ISSUE 14 Category 8

Wind tunnel results showing rotor vibratory loads reduction using higher harmonic blade pitch

HAMMOND, C. E.

AIAA 80-0667; AHS PAPER 80-66 800000 In: Structures, Structural **Dynamics,** and Materials Conference, 21st, Seattle, Wash., May 12-14, **1980,** Technical Papers. Conference sponsored by AIAA, ASME, ASCE, and **AHS. New** York, American Institute of Aeronautics and Astronautics, **Inc.,** 1980. 8 p. AA(U.S. Army, Structures Laboratory, Hampton, Va.) 8 p. Refs. 11 Jpn. 2499

The paper presents results of a wind tunnel test program using a dynamically-scaled helicopter rotor model to evaluate the use of higher harmonic blade pitch control as a means of reducing helicopter vibration levels. The test program involved the use of an adaptive automatic control system which employed Kalman filter estimation of parameters and optimal control theory. Test data are presented to show that significant reductions in the rotor vibratory vertical force and vibratory pitching moment were obtained over the range of advance ratios tested. Simultaneous reduction of the vibratory rolling moment was not achieved at all advance ratios, and the reasons behind this result remain an open issue. The wind tunnel results indicate that the higher harmonic inputs resulted in an increase in the edgewise bending moments, torsional moments, and control loads. The increased loads experienced during the test were, however, well within the design loads. The results of the test program thus indicate that active higher harmonic blade pitch control offers a viable means of achieving reduced helicopter vibration levels. (Author)

Controlled Terms: ADAPTIVE CONTROL / AUTOMATIC CONTROL / BENDING MOMENTS / HARMONIC OSCILLATION / *HELICOPTER CONTROL / *HELICOPTER TAIL ROTORS / LONGITUDINAL CONTROL / OPTIMAL CONTROL / *ROTARY STABILITY / *ROTOR AERODYNAMICS / *STRUCTURAL VIBRATION / TORSIONAL STRESS / TRANSONIC WIND TUNNELS / *VIBRATORY LOADS / *WIND TUNNEL TESTS

Reference no 164

80A34998 NASA ISSUE 14 Category 8

Multicyclic control of a helicopter rotor considering the influence of vibration, loads, and control motion

BROWN, T. J.; MCCLOUD, J. L., III

National Aeronautics and Space Administration. Ames Research Center, Moffett Field, Calif. (NC473657)

AIAA 80-0673; AHS PAPER 80-72 800000 In: Structures, Structural Dynamics, and Materials Conference, 21st, Seattle, Wash., May 12-14, 1980, Technical Papers. Part 1. (A80-34993 14-39) New York, American Institute of Aeronautics and Astronautics, Inc., 1980, p. 82-100. AB(NASA, Ames Research Center, Moffett Field, Calif.) 19 p. Refs. 7 Jpn. 2498

Weighted multiple linear regression is used to establish a transfer function matrix relationship between higher harmonic control inputs and transducer vibration outputs for a controllable twist rotor. Data used in the regression were taken from the test of a KAMAN controllable twist rotor conducted in the Ames Research Center's 40-by 80-Foot Wind Tunnel in June 1977. Optimal controls to minimize fixed system vibrational levels are calculated using linear quadratic regulatory theory with a control deflection penalty included in the performance criteria. Control sensitivity to changes in control travel, forward speed, and lift and propulsive forces is examined. It is found that the linear transfer matrix is a strong function of forward speed and a weak function of lift and propulsive force. An open-loop strategy is proposed for systems with limited control travel. (Author)

Controlled Terms: *AERODYNAMIC LOADS / DATA BASES / FEEDBACK CONTROL / *HELICOPTER CONTROL / HELICOPTER TAIL ROTORS / LINEAR EQUATIONS / *LOAD DISTRIBUTION (FORCES) / MOTION STABILITY / OPTIMAL CONTROL / REGRESSION ANALYSIS / *ROTARY WINGS / *ROTOR AERODYNAMICS / TRANSFER FUNCTIONS / *VIBRATION EFFECTS

Reference no 165

80A34995 NASA ISSUE 14 Category 8

A simple system for helicopter individual-blade-control and its application to gust alleviation

HAM, N. D.

Massachusetts Inst. of Tech., Cambridge. (MJ700802)
AIAA 80-0666; AHS PAPER 80-65 800000 In: Structures, Structural Dynamics, and Materials Conference, 21st, Seattle, Wash., May 12-14, 1980, Technical Papers. Part 1. (A80-34993 14-39) New York, American Institute of Aeronautics and Astronautics, Inc., 1980, p. 57-68. NASA-sponsored research. AA(MIT, Cambridge, Mass.) 12 p. Refs. 8 Jpn. 2497

A new, advanced type of active control for helicopters and its application to gust alleviation is described. Each blade is individually controlled in the rotating frame over a wide range of frequencies up to the sixth harmonic of rotor speed. Considerable system simplification is achieved by means of modal decomposition. It is shown both analytically and experimentally that by utilizing a tip-mounted accelerometer as a sensor in the feedback path, significant reductions in blade flapping response to a sinusoidal gust can be achieved at the gust excitation frequency as well as at super- and subharmonics of rotor speed. (Author)

Controlled Terms: ACCELEROMETERS / AIRCRAFT RELIABILITY / *GUST ALLEVIATORS / *HELICOPTER CONTROL / *HELICOPTER TAIL ROTORS / *ROTARY WINGS / *ROTOR AERODYNAMICS / ROTOR SPEED / STRUCTURAL DESIGN / STRUCTURAL VIBRATION

Reference no 166
80A27599 NASA ISSUE 10 Category 5
Operational implications of some NACA/NASA rotary wing induced velocity studies
HEYSON, H. H.
National Aeronautics and Space Administration. Langley Research Center, Langley Station, Va. (ND210491)
800200 Illinois Department of Transportation, Annual Midwest Helicopter Safety Seminar, 3rd, Joliet, Ill., Feb. 26, 27, 1980, Paper. 53 p. AA(NASA, Langley Research Center, Hampton, Va.) 53 p. Refs. 60

The purpose of the paper is to present some highlights of the broad NACA/NASA efforts throughout the years, with particular emphasis given to those results having special importance to aircraft users. Subjects covered include the rotor wake and vortex hazards, partial power descent and minimum speed for autorotation. Several aspects of ground effect are covered, including nonuniform wakes, nonlinear power and control effects in forward flight, and yaw control at near-hovering speeds. C.F.W.

Controlled Terms: BIBLIOGRAPHIES / FLOW DISTRIBUTION / FLOW VISUALIZATION / *GROUND EFFECT / *HOVERING STABILITY / *OPERATIONAL PROBLEMS / *ROTARY WINGS / *VELOCITY DISTRIBUTION / *WIND TUNNEL TESTS

Reference no 167
80A24038 NASA ISSUE 8 Category 53
The development of objective inflight performance assessment procedures
CHILDS, J. M.
790000 In: Human Factors Society, Annual Meeting, 23rd, Boston, Mass., October 29-November 1, 1979, Proceedings. (A80-24026 08-53) Santa Monica, Calif., Human Factors Society, Inc., 1979, p. 329-333. AA(Canyon Research Group, Inc., Fort Rucker, Ala.) 5 p. Refs. 5

The purpose of this research was to develop procedures for objectively evaluating Initial Entry Rotary Wing (IERW) student performance in flight. Maneuvers of the Basic Instrument phase were addressed. Descriptive inflight scoring procedures to assess absolute deviations of desired values from observed values at designated times, were developed. Desired values were determined on the basis of the rates specified in IERW training guides. Observed values were instrument indications of airspeed, altitude, or heating at those times. Four tolerance categories were incorporated into alternative six-point maneuver scoring algorithms designed to assess aircraft control precision. The criterion for acceptable proficiency was the maintenance of each sampled measure within standard IERW tolerance limits for each sampling point of a maneuver. Tests of the objective scoring procedures were conducted in the UH-1 flight simulator. (Author)

Controlled Terms: AIRCRAFT CONTROL / *AIRCRAFT MANEUVERS / ALGORITHMS / CLIMBING FLIGHT / DATA SAMPLING / FLIGHT SIMULATORS / *IN-FLIGHT MONITORING / *PILOT PERFORMANCE / *ROTARY WING AIRCRAFT / *TRAINING EVALUATION

Reference no 168
80A17716 NASA ISSUE 5 Category 5
Analysis of rotor-fuselage coupling and its effect on rotorcraft stability and response
YEN, J. G.; MCLARTY, T. T.
790000 Vertica, vol. 3, no. 3-4, 1979, p. 205-219. AB(Bell Helicopter Textron, Fort Worth, Tex.) 15 p. Refs. 13

The mode displacement, force integration, impedance and matrix displacement methods for the analysis of the dynamics of coupled helicopter rotor-fuselage systems are evaluated. The mode displacement method allows a completely coupled rotor-fuselage system to be analyzed by replacing rotor inertial couplings in the fuselage equations with stiffness couplings, and good results for rotor loads can be obtained. The force integration method is used to compute hub shears and moments by integrating dynamic and aerodynamic forces along each rotor blade, however this approach requires more computer time. Fuselage or rotor impedance, a useful concept in vibration analysis, is used to analyze rotor natural frequencies and rotor loads, and to calculate the vibration characteristics of a multi-bladed helicopter successfully and economically. The matrix displacement method systematically automates the coupling of rotating and nonrotating component equations, however requires longer computation time and exhibits poor numerical accuracy. A.L.W.

Controlled Terms: AERODYNAMIC LOADS / *AIRCRAFT STABILITY / COMPUTERIZED SIMULATION / *COUPLING / *DYNAMIC STRUCTURAL ANALYSIS / *FUSELAGES / *HELICOPTER CONTROL / HELICOPTER DESIGN / MATHEMATICAL MODELS / MATRIX METHODS / MECHANICAL IMPEDANCE / *ROTARY WINGS / STRUCTURAL VIBRATION

Reference no 169

80A15123 NASA ISSUE 3 Category 5

Examination of the flap-lag stability of rigid articulated rotor blades

KAZA, K. R. V.; KVATERNIK, R. G.

National Aeronautics and Space Administration. Lewis Research Center, Cleveland, Ohio. (ND315753)

791200 Journal of Aircraft, vol. 16, Dec. 1979, p. 876-884. AA(NASA, Lewis Research Center, Cleveland; Toledo, University, Toledo, Ohio) AB(NASA, Langley Research Center, Structures and Dynamics Div., Hampton, Va.) 9 p. Refs. 20

A critical examination of flap-lag stability of a centrally hinged, spring-restrained rigid blade in both hover and forward flight is presented. Several differences in the equations of motion for blade flap-lag stability in the existing literature are identified. A rigorous and systematic development of these equations for a rigid articulated blade in forward flight shows the existence of some linear aerodynamic coupling terms associated with blade steady-state flapping and lagging in the perturbation equations. The differences identified are shown to be associated with whether or not the lag hinge flaps with the blade. The implications of these differences on stability are examined, and it is shown that the pitch-lag coupling terms associated with a hinge arrangement in which the lag hinge flaps with the blade have a marked influence on flap-lag stability, depending on the system parameters. (Author)

Controlled Terms: AERODYNAMIC FORCES / *AERODYNAMIC STABILITY / EQUATIONS OF MOTION / *FLAPPING HINGES / FLAPS (CONTROL SURFACES) / *HOVERING STABILITY / LINEARIZATION / PERTURBATION THEORY / *RIGID ROTOR HELICOPTERS / *ROTARY WINGS / STEADY STATE

Reference no 170

80A12729 NASA ISSUE 2 Category 63

A new spectral synthesis procedure for multivariable regulators

MAYNARD, R. A.; MIELKE, R. R.; LIBERTY, S. R.; SRINATHKUMAR, S.

Old Dominion Univ., Norfolk, Va. (DS853217)

790000 NSG-1519 In: Annual Allerton Conference on Communication, Control and Computing, 16th, Monticello, Ill., October 4-6, 1978, Proceedings. (A80-12690 02-63) Urbana, Ill., University of Illinois, 1978, p. 754-761. AD(Old Dominion University, Norfolk, Va.) 8 p. Refs. 5

A method for selecting a multivariable state feedback controller that simultaneously achieves an a priori specification on closed loop eigenvalues and good mode mixing is presented. The problem is solved by projecting a desired modal matrix onto a constraint set containing the null space of the closed-loop state matrix, while assuring that the projection is in the null space. The feedback matrix follows immediately in the formulation. An example involving a helicopter hover controller is presented. (Author)

Controlled Terms: DESIGN ANALYSIS / *EIGENVALUES / EIGENVECTORS / *FEEDBACK CONTROL / *HELICOPTER CONTROL / *HOVERING STABILITY / ITERATIVE SOLUTION / *MATRICES (MATHEMATICS) / *REGULATORS / STATE VECTORS

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HELICOPTER AEROMECHANICS

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14. Abstract

The aim of this Lecture Series is to present an extensive overview of the role of aeromechanics in rotorcraft development. The influence of helicopter ability on operation of military helicopters as well as the impact of military applications on rotorcraft design ~~seem to be~~ are evident.

Following an introduction dealing with the general role of aeromechanics for rotary wing aircraft, rotorcraft aerodynamics, rotorcraft noise, dynamics of advanced rotor systems, mission requirements, handling qualities, simulation, flight testing, and advanced rotorcraft systems ~~will be covered~~.

It is considered more important to provide a general appreciation of the phenomena and to discuss the state of the art of knowledge and methods as well as new contributions to modern technology, than to discuss details of the latest achievements in various research areas.

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